CHAPTER 2. PART 27
AIRWORTHINESS STANDARDS
NORMAL CATEGORY ROTORCRAFT

SUBPART B - FLIGHT

GENERAL

AC 27.21. § 27.21 (Amendment 27-21) PROOF OF COMPLIANCE.

a. Explanation.

(1) This section provides a degree of latitude for the FAA/AUTHORITY test team in selecting the combination of tests or inspections required to demonstrate compliance with the regulations. Compliance should be shown for applicable combinations of gross weight, center of gravity, altitude, temperature, airspeed, rotor RPM, etc. Engineering tests are designed to investigate the overall capabilities and characteristics of the rotorcraft throughout its operational envelope. Testing will identify operating limitations, normal and emergency procedures, and performance information to be included in the FAA/AUTHORITY-approved portion of the flight manual. The testing must also provide a means of verifying that the rotorcraft's actual performance, structural design parameters, propulsion components, and systems operations are consistent with all certification requirements.

(2) Section 21.35 requires, in part, that the applicant show compliance with the applicable certification requirements, including flight test, prior to official FAA Type Inspection Authorization (TIA) testing. Compliance in most cases requires systematic flight testing by the applicant. After the applicant has submitted sufficient data to the FAA/AUTHORITY showing that compliance has been met, the FAA/AUTHORITY will conduct any inspections, flight, or ground tests required to verify the applicant's test results. FAA/AUTHORITY compliance may be partially determined from tests conducted by the applicant if the configuration (conformity) of the rotorcraft can be verified. Compliance may be based on the applicant's engineering data and a spot check or validation through FAA/AUTHORITY flight tests. The FAA/AUTHORITY testing should obtain validation at critical combinations of proposed flight variables if compliance cannot be inferred using engineering judgment from the combinations investigated.

(3) Performance tests include minimum operating speed (hover), takeoff and landing, climb, glide, height-velocity, and power available. Certain other performance tests, such as critical engine survey for multiengine installations, may be conducted to meet specific requirements. Detailed performance test procedures and allowable extrapolation or simulation limits are contained in the respective paragraphs in this AC.

(i) Hover tests are conducted to determine various combinations of altitude, temperature, and gross weight for both in-ground-effect (IGE) and, if required by the applicant, out-of-ground effect (OGE) conditions. From these data, the hover ceiling may be calculated.

(ii) Takeoff and landing tests are conducted to determine that a takeoff or landing can be safely executed without requiring exceptional piloting skill or favorable conditions at any approved combination of altitude, temperature, and gross weight.

(iii) For rotorcraft other than helicopters, climb tests establish the variations of rate-of-climb at the best rate-of-climb or published climb airspeed(s) at various combinations of altitude, temperature, and gross weight. For helicopter, climb tests are conducted as required to determine the best rate-of-climb speed, V_y.

(iv) Height-velocity tests are conducted to determine the boundaries of the height versus airspeed envelope from which a safe landing can be accomplished following an engine failure.

(v) Power available tests are conducted to verify the calculated installed specification engine performance model on which published performance is based.

(4) The purpose of rotorcraft stability and control tests is to verify that the rotorcraft possesses the minimum qualitative and quantitative flying qualities and handling characteristics required by the applicable regulations. In order to assess the handling qualities, standardized test procedures must be utilized and the results analyzed by accepted methods. Section 27.21(a) allows calculation and inference which includes extrapolation and simulation, whereas § 27.21(b) requires demonstration of controllability, stability, and trim. Combinations of § 27.21(a) and (b) may be used to show compliance with the operating envelope limits. Test methods and equipment are described in individual paragraphs of this advisory circular.

b. Procedures.

(1) Efforts should begin early in the certification program to provide advice and assistance to the applicant to ensure coverage of all certification requirements. The applicant should develop a comprehensive test plan which includes the required instrumentation.

(2) The tests and findings specified in paragraph AC 27.21a(3) are required of the applicant to show basic airworthiness and probable compliance with the minimum requirements specified in the applicable regulations. After these basic findings have been submitted and reviewed, a Type Inspection Authorization, or equivalent, can be issued. The FAA/AUTHORITY will develop a systematic plan to spotcheck and confirm that compliance with the regulations has been shown. The test plan will consider combinations of weight, center of gravity, and RPM and cover the range of altitude and temperature for which certification is requested.

AC 27.25. § 27.25 (Amendment 27-14) WEIGHT LIMITS.

a. Explanation.

(1) This section is definitive and specifies criteria for establishing maximum and minimum certificating weights. These weights may be based on those selected by the applicant, design requirements, or the limits for which compliance with all applicable flight requirements has been shown.

(2) Typical requirements that may establish the maximum and minimum weight limits include:

(i) Maximum: Structural limits, performance requirements, stability, and controllability requirements.

(ii) Minimum: Autorotative rotor RPM, stability, and controllability requirements.

(3) Jettisonable External Cargo.

(i) Section 27.25(c) was added by Amendment 27-11 to provide a basis for approving an increased gross weight that would be an external jettisonable load. Section 27.865, "External load attaching means, " includes hoist and hook design features for the load attaching devices that were added to Part 27 but removed from § 133.43. Part 133, "Rotorcraft External-Load Operations," was also amended (Amendment 133-5) concurrently to complement the changes to Parts 27 and 29.

(ii) Approvals under the policy in Review Cases Nos. 37 and 55 of FAA Order 8110.6 were no longer necessary. These review cases concerned the policy/standards for external cargo configurations using a cargo hook whenever the standard limitations were exceeded. If the standard limitations were not exceeded, external cargo hooks and hoists and external cargo configuration approvals could be made under Part 133, Subpart D, prior to Amendment 133-5.

(iii) In the preamble of Amendment 27-11 (Proposal 2-99, 41 FR 55454; December 20, 1976) the agency stated that "...§ 27.25(c) and § 29.25(c) are intended to provide only a total weight standard for approving the rotorcraft structure for operation under Part 133." The policy in Review Case No. 55 also indicates the powerplant or propulsion system is also subject to evaluation for the increased weight. As indicated in § 27.865, fatigue substantiation of the external cargo attaching means is not required. The rotorcraft structure, rotors, etc., are only subject to fatigue evaluation under § 27.571 whenever the standard structural limitations are exceeded (Review Case No. 55).

(iv) Whether or not the standard limitations are exceeded, the flight characteristics evaluations/standards of § 133.41 are appropriate even for engineering approval. Section 133.41 is also appropriate for the individual operator to obtain his operating certificate. The operator may use an FAA/AUTHORITY approved RFM supplement to prepare his own rotorcraft load combination flight manual required by § 133.47.

b. Procedures.

(1) It may not be possible to demonstrate quantitatively all the flight requirements at the minimum weight because of test instrumentation requirements. The test team must ensure that the rotorcraft complies with the applicable requirements at the lowest permissible flying weight. This evaluation may be done qualitatively with the test instrumentation removed and with minimum crewmembers if no critical areas exist or are anticipated. Additionally, reasonable extrapolation is permitted. However, if critical areas at minimum flying weights are apparent, extrapolation should not be permitted.

(2) Whenever a gross weight increase under § 27.25(c) is requested, a TIA evaluation is necessary to evaluate the new limitations and ensure that § 133.41 for typical or representative cargo weights and/or shapes (or density) is satisfactory. All possible combinations of weights and shapes are not evaluated. The representative configurations may be noted in the RFM or RFM Supplement for the operator's information. Sections 133.41 and 133.47 must be satisfied by the individual operator for the particular case at hand. The approved RFM or RFM Supplement should provide the necessary limitations and any other information about the representative cargo configurations evaluated. Section 133.41 also permits the operator to obtain approval of additional and unique cargo configurations provided approved limitations are observed. Paragraph AC 27.1581 concerns the RFM and its contents.

(3) See AC 29-2C, Certification of Transport Category Rotorcraft, paragraph AC 29.571, concerning § 29.571, for fatigue substantiation and external cargo considerations that apply to § 27.571 as well.

(4) Refer to AC 133-1A, Rotorcraft External-Load Operations in Accordance with FAR Part 133, for further information on airworthiness and flight manual policy for operators.

AC 27.27. § 27.27 (Amendment 27-2) CENTER OF GRAVITY LIMITS.

a. Explanation.

(1) This regulation is definitive and requires that the center of gravity limits be defined. Proof of compliance with all applicable flight requirements is required within the range of established CG's. Along with the longitudinal CG limits, the lateral CG limits should either be established or determined to be not critical.

(2) Ballast is usually carried during the flight test program to investigate the approved gross weight/center of gravity limits. Lead is the most commonly used form of ballast during rotorcraft flight testing although other types of ballast, such as water, may serve just as well. Water may have the added benefit of being jettisonable during critical flight test conditions. Care must be taken regarding the location of ballast. The strength of the supporting structures should be adequate to support such ballast during the flight loads that may be imposed during a particular test and for the ultimate inertia forces of § 29.561(b)(3). Of critical importance is the method of securing the ballast to the desired locations. To avoid any undesired in-flight movements of the ballast, a positive method of constraint is mandatory. The flight test crews should also visually verify the amount, location, and integrity of the ballast. The effects of mass moment of inertia on the flight characteristics due to the ballast locations should also be considered. The mass moment of inertia of the test rotorcraft should, to the extent possible, be the same as that expected in normal, approved loadings, especially during tests involving dynamic inputs.

b. Procedures.

(1) Center of gravity locations and limits are of prime importance to rotorcraft stability and safety in flight. The primary concern is establishment of the longitudinal center of gravity limits. Lateral center of gravity limits with respect to longitudinal center of gravity limits are also important. The design of the rotorcraft is usually such that approximate lateral symmetry exists. This lateral symmetry can be upset by numerous probable lateral loadings possibly resulting in the necessity to establish lateral center of gravity limits. Stability and control characteristics may be seriously affected by loading outside the established center of gravity limits. The established center of gravity limits must be that as fuel is consumed, it is possible for the rotorcraft to remain within the established limits by acceptable loading and/or operating instructions.

(2) Structural limits may restrict the maximum forward longitudinal center of gravity limits. However, in most cases it is the maximum value established wherein adequate low speed control power exists to meet such requirements as § 27.143(c). Likewise, the maximum aft center of gravity limit may be a “structural limit,” but it usually is determined during flight test after the rotorcraft’s handling qualities tests have been conducted. Flight tests may reduce the “structural limit” CG envelope, but flight tests alone should not be used to expand the “structural limit.” Additional items which may influence the maximum aft center of gravity limits may be malfunctions of automatic stabilization equipment, excessive rotorcraft attitudes during critical phases of flight, or adequate control power to compensate for an engine failure.

(3) Lateral center of gravity limits have become more critical because of the ever increasing utilization of the rotorcraft for such things as unusual and unsymmetric lateral loads, both internal and external. Maximum allowable lateral center of gravity limits have also influenced the results of the unusable fuel determination.

(4) In summary, it is of prime importance that longitudinal and lateral center of gravity limits be determined so that unsafe conditions do not exist within the approved altitude, airspeed, ambient temperature, gross weight, and rotor RPM ranges. All relevant malfunctions must be considered.

AC 27.29. § 27.29 (Amendment 27-14) EMPTY WEIGHT AND CORRESPONDING CENTER OF GRAVITY.

a. Explanation. The empty weight of the rotorcraft consists of the airframe, engines, and all items of operating equipment that have fixed locations and are permanently installed in the aircraft. It includes fixed ballast, unusable fuel, and full operating fluids except water intended for injection in the engines.

(1) Fixed ballast refers to ballast that is made a permanent part of the rotorcraft as a means of controlling the empty weight CG.

(2) Compliance with paragraph (b) of § 27.29 is accomplished by the use of an equipment list which defines the installed equipment at the time of weighing and the weight arm and moment of the equipment.

b. Procedures.

(1) Determination of the empty weight and corresponding center of gravity is primarily the responsibility of the manufacturing inspector. This determination is normally made on the production rotorcraft rather than the prototype. If the manufacturer wishes to avoid the necessity of weighing each production rotorcraft and he has been issued a production certificate, he may make a detailed proposal defining the procedures he will use to establish an empty weight and CG When his proposal is approved, he will weigh the first five to ten production rotorcraft and show that the rotorcraft will be within ±1 percent on empty weight and ±0.2 inches on CG After this procedure is established, the empty weight and CG may be computed except that at regular intervals, a rotorcraft will be weighed to ensure the tolerances are still being maintained; e.g., one in ten rotorcraft.

(2) For prototype and modified rotorcraft, it is only necessary to establish a known basic weight and CG position (by weighing) from which the extremes of weight and CG travel required by the test program may be calculated. See AC 91-23 (Pilots Weight and Balance Handbook) for a sample weight and balance procedure.

(3) The weight and balance should be recalculated if a modification (or series of modifications) to the rotorcraft results in a significant change to the empty weight. Additionally, this change in empty weight should be reflected with the weight and balance information contained in the Rotorcraft Flight Manual (RFM) or Rotorcraft Flight Manual Supplement (RFMS).

c. Ballast Loading and Type.

(1) Ballast loading of the rotorcraft can be accomplished in any manner to achieve a specific CG location. It is acceptable for such ballast to be mounted outside the physical confines of the rotorcraft if the flight test objectives are not affected by this arrangement. In flight test work, loading problems will occasionally be encountered in which it will be difficult to obtain the desired CG limits. Such cases may require loading in engine compartments or other places not designed for load carrying. When this condition is necessary, care should be taken to ensure that local structural stresses are not exceeded or that the rotorcraft flight characteristics are not changed due to increased moments of inertia by attaching the ballast to extreme CG locations which may not be designed for the added weight.

(2) There are basically two types of ballast that may be used in loading. They are solids or liquid. The solids are usually high density materials such as lead while the liquid usually used is water. In critical tests, the ballast may be loaded in a manner so that disposal in flight can be accomplished. In any case, the load should be securely attached in its loaded position so shifting or interference with safety of flight will not result.

AC 27.31. § 27.31 REMOVABLE BALLAST.

a. Explanation. This regulation provides the option of using removable ballast to obtain desired center of gravity locations to determine compliance with the flight requirement of this Part. Fixed ballast used for flight operations after type certification must be documented in the type design data. Removable ballast is used primarily on small rotorcraft to control the CG with different passenger loadings although this regulation does not permit its use on transport rotorcraft. If removable ballast is used, the rotorcraft flight manual must include instructions regarding its use and limitations.

b. Procedures. None

AC 27.33. § 27.33 (Amendment 27-14) MAIN ROTOR SPEED AND PITCH LIMITS.

a. Explanation.

(1) General. This section requires the establishment of power-on and power-off main rotor speed limits and the requirements for low rotor speed warning.

(2) Power-On. The power-on limits should be sufficient to maintain the rotor speed within these limits during any appropriate maneuver expected to be encountered in normal operations throughout the flight envelope for which certification is requested. In the past a minimum power-on range of approximately 3 percent has been required due to engine governor and engine operating characteristics. With the introduction of advanced engines and electronic engine controls, there may not be a need for a range. One fixed value may suffice. If substantiated, transient power-on values may also be acceptable.

(3) Power-Off. The power-off rotor speed limits should be sufficient to encompass the rotor speeds encountered during normal autorotative maneuvers except for final landing phase (touchdown) for which rotor RPM may be lower than the minimum transient limit for flight, provided stress limits are not exceeded. The limits should also be sufficient to cover the ranges of airspeed, weight, and altitudes for which certification is requested. It is not the intent of the rule to require the minimum and maximum limit values in conjunction with extremes such as maximum/minimum weights and/or high altitude. The minimum and maximum rotor speed requirements should be thoroughly evaluated at normal operating environment; i.e., at altitudes from approximately sea level to 10,000 feet, temperatures not at extremes, and weights as necessary for other tests and as required to readily establish the limit rotor speeds. Spot checks of the autorotative requirements should be made at the extremes of the flight envelope and environmental conditions during normal tests at those conditions. Under conditions where high autorotative rotor speeds may be encountered, it is acceptable for the pilot to adjust the controls to prevent overspeeding of the rotor. At light weight combined with low altitudes and extremely cold temperatures, the normal low pitch setting may not be sufficient to maintain autorotational rotor speed values within limits. If this occurs, the manufacturer may elect to adjust the low pitch stops as a maintenance procedure at extreme ambient conditions provided the flight and maintenance manuals clearly present the rigging requirements and procedures. There must be sufficient “overlap” of ambient conditions between configurations such that rerigging is not required whenever ambient temperature and surface elevation change slightly. Any downrigging of the low pitch stop must continue to ensure adequate clearance between controls and other rotorcraft structure and should be evaluated during flight test. Both the power-on and power-off limits may also be established by encountering critical flapping limits in some approved flight conditions such as high airspeed or sideward flight.

(4) Low Speed Warning. If it is possible under expected operating conditions for the rotor speed to fall below the minimum approved values, the requirement exists for a low rotor speed warning. This warning is required on all single-engine rotorcraft and on multiengine rotorcraft where there is not an automatic increase in remaining engine(s) power output upon failure of an engine. Although not required by the rule, essentially all of today’s multiengine rotorcraft have a low rotor speed warning system installed. If the minimum power-on and power-off rotor speed limits are different, the warning signal should be at the higher speed, normally the power-on minimum rotor speed. One type of rotorcraft has a warning system cutout if the collective is full down, and other types have other warnings on the engine speed to indicate engine failure. All of these related warning systems must be evaluated with emphasis on ensuring adequate rotor speed.

b. Determination and Testing. Refer to paragraph AC 27.1509 (§ 27.1509) for additional information on determining and testing rotor limits.

SUBPART B - FLIGHT

PERFORMANCE

AC 27.45. § 27.45 (Amendment 27-21) PERFORMANCE--GENERAL.

a. Explanation.

(1) Section 45 of Part 27 lists some of the rules and standards under which the performance requirements are to be met. This paragraph will provide general guidelines that may be used throughout a flight test program. It is impossible to find ideal test conditions and there are many variables which affect the flight test results that must be taken into account. Some of these variables are wind, temperature, altitude, humidity, rotorcraft weight, power, rotor RPM, center of gravity, etc. A thorough knowledge of the testing procedures and data reduction methods is essential and good engineering judgment must be used to determine acceptable test conditions. The test results should be analyzed and expanded by approved methodology within the guidelines of this paragraph.

(2) Performance should be based on approved engine power as determined in paragraph b(5) below and not on any transient limits. Approved transient limits are basically for inadvertent overshoots of approved operational limits and any sustained operation in these transient limit areas usually requires some form of special maintenance. However, for such demonstrations as landing procedure demonstration and height-velocity (HV) determination, low rotor speeds (within approved limits) have been authorized. Such transients, if authorized, must be flight evaluated.

(3) Where variations in the parameter on which a tolerance is allowed will have an appreciable effect on the test, the results should be corrected to the standard value of the parameter; otherwise, no correction is necessary.

b. Procedures.

(1) Winds for Testing.

(i) Allowable wind conditions will vary with the type of test and will also be different for different types and gross weight rotorcraft. For example, higher winds can usually be tolerated for takeoff and landing tests than for hover performance. Higher winds can sometimes be tolerated during hover performance testing on rotorcraft with high rotor downwash velocities. Generally, unless the effects of wind on hover performance tests can be determined and/or accounted for, hover performance testing should be conducted in winds of 3 knots or less.

(ii) In-ground-effect controllability and maneuverability testing should be conducted in surface winds of less than 5 knots, or when higher steady wind conditions exist, with a maximum gust spread of 5 knots.

(iii) As can be seen from the foregoing, there is no such thing as an exact allowable wind for a particular test or rotorcraft. The flight test team must decide on the allowable wind for each condition based on all available information and their engineering judgment. The following summary of allowable wind conditions is given for general guidance only:

(A) Hover performance - 0 to 3 knots.

(B) Height-velocity - 0 to 3 knots.

(C) IGE controllability and maneuverability - 0 to 5 knots.

(iv) A means should be provided to measure the wind velocity, direction, and ambient air temperature at the rotor height for any particular tests.

(2) Altitude Effects. Using FAA/AUTHORITY-approved methodology, hover performance may be extrapolated and/or interpolated from test data up to a maximum of ±4,000 feet. Experience has shown that IGE handling qualities, height-velocity, and engine operating characteristics should not be extrapolated higher than approximately 2,000 feet density altitude from the test altitude. Cruise stability/controllability tests should be evaluated at least at two different altitudes, the lowest practical altitude and approximately the highest cruise altitude requested for approval. This can allow an interpolation of approximately 10,000 feet. As in all testing, extrapolation and/or interpolation should only be considered if all available information and engineering judgment indicate that regulatory compliance can be met at the untested conditions.

(3) Altitude Limitations.

(i) Explanation.

(A) Two altitudes are normally presented in the RFM to define the operating envelope of a rotorcraft;

- Maximum operating altitude, and
- Maximum takeoff and landing altitude.

(B) Maximum operating altitude is an operating limitation required by § 27.1527 and delineates the maximum altitude to which operation is allowed. This altitude normally constitutes the maximum cruise or en route altitude.

(C) Maximum takeoff and landing altitude is the hover in-ground-effect (IGE) ceiling for a rotorcraft as described in § 27.73. The hover ceiling and any information pertinent to takeoff and landing are presented in the performance information section of the RFM. For rotorcraft certified to CAR 6, Amendment 6-7 or any amendment of FAR 27, a hover ceiling may not be presented above the altitude at which H-V and IGE controllability tests were conducted plus allowable extrapolation, unless that extrapolated altitude is at least 7,000 feet. If the applicant elects to demonstrate these tests to an altitude below 7,000 feet, then that altitude is the maximum takeoff and landing altitude of the rotorcraft. The maximum takeoff and landing altitude may be coincident with, but never above the maximum operating altitude limitation. Takeoff and landing and hover ceiling data and presentation requirements are presented in §§ 27.51, 27.73 and 27.1587.

(ii) Procedures.

(A) In establishing the maximum takeoff and landing altitude, the following tests are normally required:

(1) Takeoff (§27.51)

(2) Climb (§§ 27.65 and 27.67)

(3) Performance at minimum operating speed (§ 27.73)

(4) Landing (§ 27.75)

(5) Limiting height-speed envelope (§ 27.79)

(6) IGE controllability (§ 27.143c)

(7) Cooling (§§ 27.1041, 27.1043 and 27.1045)

(8) Engine operating characteristics (§ 27.939)

Specific guidance on test methodology and data requirements is provided in applicable paragraphs of this AC.

(B) As detailed in subparagraph b(2) above, the maximum allowable extrapolation of H-V, IGE controllability and engine operating characteristics is ±2,000 feet. Therefore, the maximum takeoff and landing altitude presented in the RFM is not normally more than 2,000 feet above the density altitude experienced at the high altitude test site, or for CAR 6, Amendment 6-7 and subsequent, unless test results were demonstrated to at least 7,000 feet.

(C) If IGE controllability is demonstrated to at least 17 knots of wind at 7,000 feet, hover capability above 7,000 feet may be presented provided that the maximum demonstrated safe wind for takeoff and landing above 7,000 feet is specified in the RFM.

(D) The requirements for data collection and presentation in the RFM vary depending upon the certification basis of the rotorcraft. These requirements are presented by regulation and amendment in figures AC 27.45-1 and AC 27.45-2.

(E) The maximum takeoff and landing altitude may be extrapolated no greater than the values given in paragraph b(2) and not above the lowest limiting altitude resulting from the requirements of subparagraph A of this paragraph.

(4) Temperature Effects.

(i) Background.

(A) In the past, approved analyses were frequently accepted for determining the extreme temperature effects on performance and flight characteristics. With the introduction of newer, higher performance rotorcraft, advanced rotor blade designs, higher airspeeds, and higher blade tip Mach numbers, the previous methods have proven to be insufficient. Therefore, the performance and flight characteristics should be validated at extreme temperatures; however, analysis may be permitted if a suitable methodology is demonstrated.

(B) Various FAA/AUTHORITY cold weather programs have verified that rotorcraft can be affected by cold temperature in both the performance and flying qualities areas. Hot temperature conditions, although not shown to be as critical for flying qualities, should be given consideration.

(C) Additionally, design deficiencies surfaced when the rotorcraft were exposed to temperature extremes and some of these difficulties were severe enough to require the redesign of equipment and/or materials. Therefore, to satisfy § 27.1309(a), the applicant needs to substantiate the total rotorcraft throughout the foreseeable range of operating temperatures.

(ii) Procedures.

(A) The FAA/AUTHORITY is responsible for verifying the effects of temperature on performance and handling characteristics. A limited flight verification, if necessary, could include spot checks of hover performance, IGE controllability, vibration, simulated power failure, static stability, height-velocity, V_{NE}/V_D evaluations, ground resonance, etc. In addition, systems should be evaluated to determine satisfactory operations.

(B) Extrapolation of test data should only be allowed if the applicant's predicted or calculated data is verified by actual test, but in any case extreme caution should be used for extrapolations that are 10° C below or 20° C above those values tested.

(5) Engine Power - Turboshaft Engine.

(i) Background.

(A) The purpose of rotorcraft performance flight testing is to obtain accurate quantitative flight test performance data to provide flight manual information.

(B) Flight tests are designed to investigate the overall performance capabilities of the rotorcraft throughout its operating envelope. This testing furnishes information to be included in the flight manual and provides a means of validating the predicted performance of the rotorcraft with a minimum installed specification engine.

(C) The power used to complete the flight manual performance must be based on power values no greater than that available from the minimum uninstalled specification engine after it is corrected for installation losses. A minimum uninstalled specification engine is one that, on a test stand under conditions specified by the engine manufacturer, will produce the certificated power at specification temperatures and/or speeds. The specification values may be either a rating or limit. Some engine manufacturers certify an engine to a specified power at a particular engine temperature or speed rating with higher allowable limits. The limit is the maximum value the installed engine is allowed in order to develop the specification power. Prior to installation of each engine in a rotorcraft, the performance is measured by the engine manufacturer. This is done by making a static test run in a test cell and referring the results to standard day, sea level conditions. The performance parameters obtained are presented as uninstalled engine characteristics on a test log sheet. This is commonly referred to as a “final run sheet.” Figure AC 27.45-3 compares a typical engine to one the manufacturer has certified as a minimum uninstalled certified engine.

(D) After engine certification, the engine manufacturer is responsible to ascertain that each engine delivered will produce, as a minimum, the certified power without exceeding specification operating values; therefore, a “final run sheet” is created for every engine produced. Additionally, if needed, arrangements can usually be made with the engine manufacturer to obtain a torque system calibration for individual engines. This will further optimize the accuracy of the engines used in the flight test program. The engine manufacturer will also provide predicted uninstalled power available for the various power ratings. This information may be derived from an engine computer “card deck” and from charts and tables in the engine detail installation manual. These data also provide engine performance for the range of altitudes and temperatures approved for the engine and include methods for correcting this performance for installation effects. The parameters contained in a typical “card deck” are plotted for one engine rating in figure AC 27.45-4.

(E) Several power losses may be associated with installing an engine in a rotorcraft. Typical losses are air inlet losses, gear losses, air exhaust losses, and powered accessory losses such as electrical generators. Additional flight manual performance considerations are the torque indicating system accuracy and torque needle split. The predicted uninstalled power available engine characteristics cannot be assumed to be the actual power available after the engine is installed in the rotorcraft because this procedure would neglect the installation power losses. It is necessary to know the installation losses in order to determine the flight manual performance. Installation losses are reflected reductions in available power resulting from being installed in a rotorcraft. These losses usually consist of those incurred due to engine inlet and/or exhaust design. The rotorcraft manufacturer conducts tests to confirm the installed specification engine power available on which published performance is based. The specific methods used vary widely between manufacturers but usually include some combination of ground and flight tests. Figure AC 27.45-5 is a typical installed power available chart for one set of conditions.

(F) The installed power available is, in most cases, lower than obtained on a test stand. This is especially true at lower airspeeds where exhaust reingestion may occur and there are changes in airflow routing. The rotorcraft manufacturer may elect to determine the installation losses for different flight conditions to take any airspeed advantages. This is acceptable if, for example, the hover performance is based on the actual power available from an installed minimum specification engine <u>in a hover</u>. Likewise, it is permissible for the rotorcraft manufacturer to determine his climb performance based on the actual power available from an installed minimum specification engine at the published climb airspeed. This will allow the manufacturer to take advantage of, for example, increased inlet efficiency.

(ii) <u>Procedures.</u>

(A) The installed minimum specification engine power output has been predicted and calculated for various flight conditions. It is imperative that the predicted values be verified by actual flight test. The flight test involves obtaining engine performance measurements at various power settings, altitudes, and ambient temperatures. The data should be obtained at the actual flight condition for which the performance is to be presented (i.e., hover, climb, or cruise).

(B) Following a power increase, engine temperature and/or RPM can significantly decrease for a period of time as torque is held constant. Said another way, torque will increase if RPM and/or temperature are held constant. This is a characteristic typical of turbine engines due largely to expansion of turbine blades and reduced clearances in the engine. Some engines may show a temperature increase at constant power due to engine or temperature sensing system peculiarities. An engine will usually establish a stabilized relationship of power parameters in approximately 2 or 3 minutes. For this reason, the following procedure should be used when obtaining in-flight engine data.

(<u>1</u>) To determine the takeoff and 2 ½-minute values, first stabilize the engine at a low power setting. After stabilization, rapidly increase the power demand to takeoff and/or 2 ½-minute power levels. Record the engine parameters as soon as the specification torque, temperature, or speed is attained. Care must be taken not to exceed a limit. These readings should be obtained approximately 15 seconds after power is initially applied.

(2) To determine the 30-minute and/or maximum continuous power values, approximately 2 to 3 minutes of stabilization time after power is increased is generally used, but up to 5 minutes stabilization time is allowed. The reason for the different procedures is when a pilot requires takeoff or 2 ½-minute power values he is in a critical flight condition and does not have the luxury of waiting for the engine(s) to produce rated power. Stabilization time is allowed for the maximum continuous and 30-minute ratings because these values are not associated with flight conditions for which power is needed immediately.

(C) The in-flight measurements recorded with the engine(s) on the flight test rotorcraft must be corrected downward if the test engine is above minimum specification and corrected upward for a test engine that is below minimum specification. This correction is necessary to verify that a minimum specification engine installed on a production rotorcraft is capable of producing the power values used to compute the flight manual performance without exceeding any engine limit. In addition, if the production rotorcraft's power measurement devices have significant (greater than 3 percent) power error, this error must be accounted for in a conservative manner.

(D) On multiengine rotorcraft, the engine location may result in different installation losses between engines. If this condition exists, multiengine performance should be based on the total power available after considering the different installation losses and with minimum specification engines installed. One-engine-inoperative performance must be based on the loss of the engine which has the lowest installation losses. Additionally, the power losses due to such items as accessory bleed air, particle separators, engine driven accessories, etc., must be accounted for accordingly.

(E) Power available data should be obtained throughout the test program at various ambient conditions. Some engines have devices which restrict the mechanical N_G speed to a constant corrected speed at cold temperatures. Others may limit power to a fuel flow value which would be encountered only at certain ambients. Others may limit by torque limiting devices. Therefore, power available data should be obtained at various ambients to verify that all limiting devices are functioning properly and have not been affected by the installation.

(F) Through use, turbine engine power capabilities decrease with time. This is called engine deterioration. Deterioration is largely a function of the particular engine design, the manner, and the environment in which the engine is operated. There is a need, therefore, to provide a method which can be used in service to periodically determine the level of engine deterioration. A power assurance curve is usually provided to allow the flightcrew to know the power producing capabilities of any engine. A power assurance check is a check of the engine(s) which will determine that the engine(s) can produce the power required to achieve flight manual performance. This check does not have to be done at maximum engine power. Figure AC 27.45-6 is a typical power assurance curve for an installed engine showing minimum acceptable torque which assures that power is available to meet the rotorcraft flight manual performance. Some power assurance curves have maximum allowable N_G limits that must not be exceeded for a given torque value. An in-flight power assurance check may be used in addition to the pretakeoff check. The validation of either check must be done by the methodology used to determine the installed minimum specification engine power available. For the in-flight power assurance check there must be full accountability for increased efficiency due to such items as inlet ram recovery, absence of exhaust reingestion, etc. A power assurance check done statically and one conducted in-flight must yield the same torque margin(s). An engine may pass power assurance at low power but still may not be capable of producing the rated power values. This occurs when the curve of corrected power and corrected temperature for the engine intersects the minimum uninstalled specification engine curve. If this condition exists, the entire power assurance and power available information may need to be reestablished.

(6) Deteriorated Engine Power - Turboshaft Engine.

(i) Background.

(A) A specific engine model may have been certificated for operation with power which has “normally” deteriorated below specification. This “normal” deterioration refers to a gradual loss in engine performance, possibly caused by compressor erosion, as opposed to a sudden performance loss which may be due to mechanical damage. The application for deteriorated engine power should not be confused with the installed mechanical engine derating which is frequently used to match transmission and engine power capabilities.

(B) The use of deteriorated power is intended to allow continued operations with an engine which is serviceable and structurally sound, although aircraft performance may be depreciated. The useful life of the engine may, therefore, be extended at a dollar savings to the operator.

(C) Although installed performance is the primary topic in this discussion, considerations must be given to other operational characteristics and systems which may be affected by deteriorated engine power. These include:

(1) Engine characteristics (§ 27.939). Surge margin, engine response, and air-restart capability might be affected and should be addressed, but flight testing may not be required depending on the individual engine/aircraft installation and fuel scheduling mechanism.

(2) Performance of customer bleed air systems may be degraded slightly. No problem would be anticipated unless certain items within the system depend on a critical P_C for their function.

(3) The maximum attainable gas producer speed, and thus power available under certain ambients, may be affected if P_C is an input to the fuel scheduling mechanism.

(4) Systems for surge protection which schedule on P_C such as bleed valves, flow fences, bleed bands, and variable inlet guide vanes may be influenced. The effect would normally be negligible unless when installed, the installation losses, combined with reduced P_C because of deterioration, would cause the bleed device to open and reduce power at any one of the engine ratings.

(ii) Procedures.

(A) The need for flight tests to verify predicted power available with deteriorated engines depends on the scope of testing which occurred during initial certification. If the original rotorcraft certification included flight testing as described in paragraph (5) (engine power-turboshaft engines) herein for validation of power available, the need for a demonstration with deteriorated engines is greatly diminished and perhaps eliminated.

(B) If flight testing to verify deteriorated engine power available is deemed necessary, the procedure used would be the same as that described in paragraph (5) (engine power-turboshaft engines), except that the data would be corrected downward to a deteriorated engine runline. Efforts should concentrate on obtaining data in areas of the operational envelope where maximum gas producer speed is likely to be attained, or where bleed valves or other devices which schedule on gas producer discharge pressure are likely to function. On many installations maximum gas producer speed will occur with cold temperatures and high altitudes; bleed valves and other devices which schedule on gas producer discharge pressure are most likely to function and reduce power on a hot day at low altitude.

(C) The adjustments to the normal power assurance check procedures for deteriorated engines will be influenced by the preferences of the aircraft manufacturer and by any special stipulations of the engine certification established as a condition for the engine to remain in service when below specification. Possibly, more stringent and more complicated engine monitoring procedures will be introduced when allowing the use of deteriorated power; for example, an in-flight trend monitoring program with the associated bookkeeping duties may be required. Such an in-flight procedure must be evaluated by flight tests as described in paragraph (5) (engine power-turboshaft engines) herein. Normally, however, the manufacturer would be expected to present a modification, or extension of the power assurance procedure already in place for the specification engine, which could eliminate the need for flight test evaluation.

AC 27.45A. § 27.45 (Amendment 27-21) GENERAL.

a. Explanation. Amendment 27-21 adds § 27.45(f) to the regulation. This section establishes the requirement for furnishing power assurance information for turbine powered rotorcraft. This information is to provide the pilot a means of determining, prior to takeoff, that each engine will produce the power necessary to achieve the performance presented in the rotorcraft flight manual (RFM).

b. Procedures. All of the policy material pertaining to this section remains in effect. In addition, the power assurance information included in the RFM should be verified. Although this requirement is normally met with a power assurance curve, other methods of compliance may be proposed.

AC 27.45B. § 27.45 (Amendment 27-29) GENERAL.

a. Explanation. Amendment 27-29 added the requirements for certification of 30-second/2-minute One Engine Inoperative (OEI) power ratings. For rotorcraft approved for the use of 30-second/2-minute OEI, partial power checks currently accomplished with approved power assurance procedures for lower power levels may not be sufficient to guarantee the ability to achieve the 30-second power level.

b. Procedures. Information provided in Appendix 1 of this AC includes guidance material on power assurance procedures to assure that the OEI power level can be achieved

CERTIFICATION BASIS

		FAR 27			CAR 6			CAR 06
Rqmts		**27-Amdt. 21**	**27-Amdt. 2**	**Original**	**6-Amdt. 7**	**6-Amdt. 4**	**Original**	**Original**
H-V Ref. 27.25 27.79 27.1519 27.1587 6.116 6.741 6.743	**Test Conditions**	1. MGW Sea Level 2. Max. OGE wt. Lesser of: a. Max. alt. cap. b. 7000' Hd	1. MGW Sea Level 2. Wt. selected by applicant at lessor of: a. Max. alt. cap. b. 7000' Hd	1. MGW Sea Level 2. Wt. selected by applicant at lessor of: a. Max. alt. cap. b. 7000' Hd	1. MGW Sea Level 2. Wt. selected by applicant at lessor of: a. Max. alt. cap. b. 7000' Hd	1. MGW Sea Level 2. Wt. selected by applicant max. t.o. and ldg. alt.	1. MGW Sea Level 2. Wt. selected by applicant at max. t.o. and ldg. alt.	1. No specific wt. and alt. requirements.
	RFM	3. H-V is perf. Info. 4. Max. alt. For which H-V is valid	3. H-V is perf. info. 4. If H-V wt is less than IGE wt., wt. becomes limitation for t.o./ldg.	3. H-V is perf. info.	3. H-V is perf. info.	3. H-V is perf. info.	3. H-V is operating limitation.	H-V is operating limitation.
AC 27-1B Para AC 27.45 & AC 27.51	**Remarks**	5. If H-V is less than OGE wt., H-V wt. becomes limit. 6. Applicant is encouraged to demo H-V to WAT limits. 7. Hover data may be shown above 7000' if H-V & IGE are demo'd to 7000'						

FIGURE AC 27.45-1 H-V Requirements

CERTIFICATION BASIS

		FAR 27			CAR 06
Rqmts		**27-Amdt. 21**	**Original**	**Original**	**Original**
IGE CONTROL Ref. 27.25 27.143 27.1587 AC 27-1B Para AC 27.45 & AC 27.143	**Test Conditions**	1. MGW Sea Level 2. Max. IGE wt. Lesser of: a. Max. alt. cap. b. 7000' Hd 3. Critical CG 4. Critical Rotor RPM 5. Wind of not less than 17 kts.	1. MGW Sea Level 2. Wt. selected by applicant to max. t.o. and ldg. alt. 3. Critical CG 4. Critical Rotor RPM 5. Critical wt. 6. Wind of not less than 20 mph	1. MGW Sea Level 2. Max. approved wt. for t.o./ldg at alt above sea level. 3. Critical CG 4. Critical Rotor RPM 5. Wind not less than 20 mph.	1. No specific requirement.
	RFM	6. Max. safe wind is perf. info.	7. Max. safe wind is perf. info.	6. Max. safe wind is perf. info.	
	Remarks	7. If 17 kts. wind demo'd to alt. less than 7000', a corresponding WAT limit must be established.			

FIGURE AC 27.45-2 IGE CONTROLLABILITY REQUIREMENTS

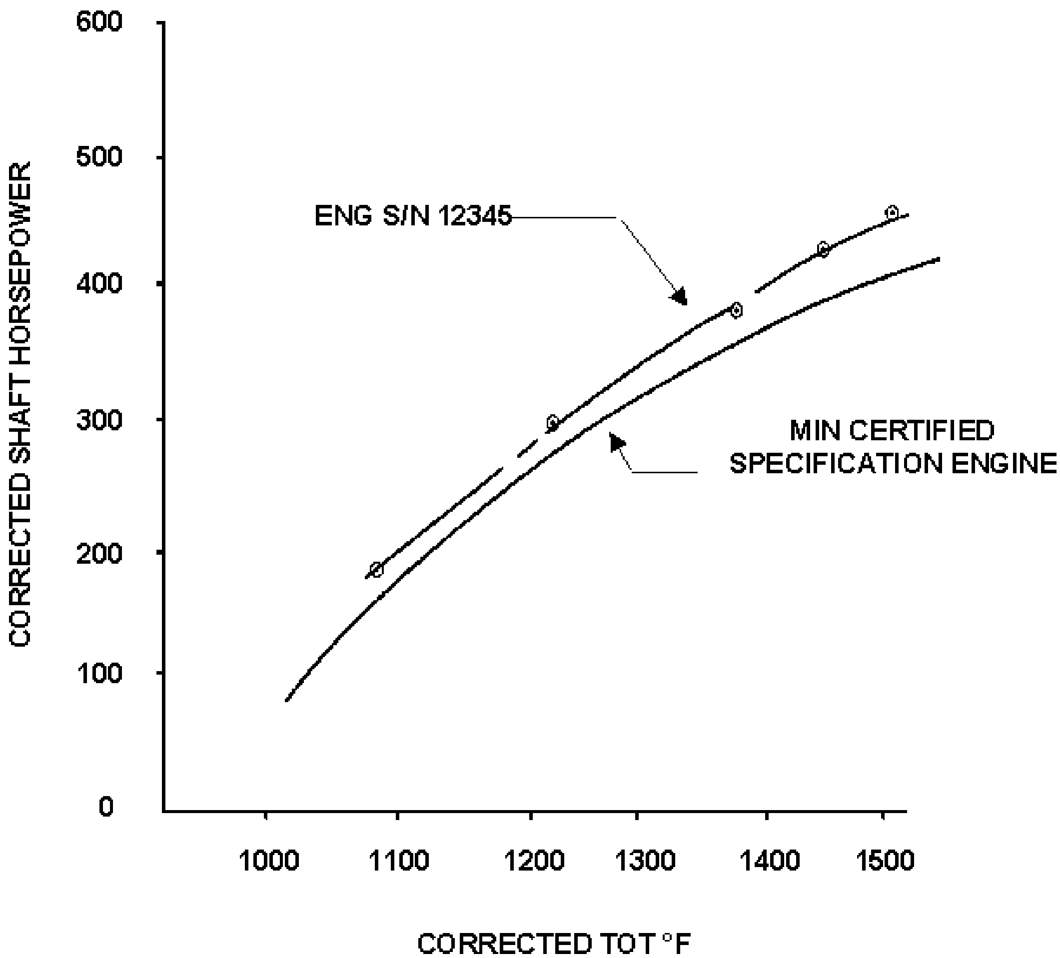


FIGURE AC 27.45-3 SHAFT HORSEPOWER VS TURBINE OUTLET TEMPERATURE - SEA LEVEL STANDARD DAY

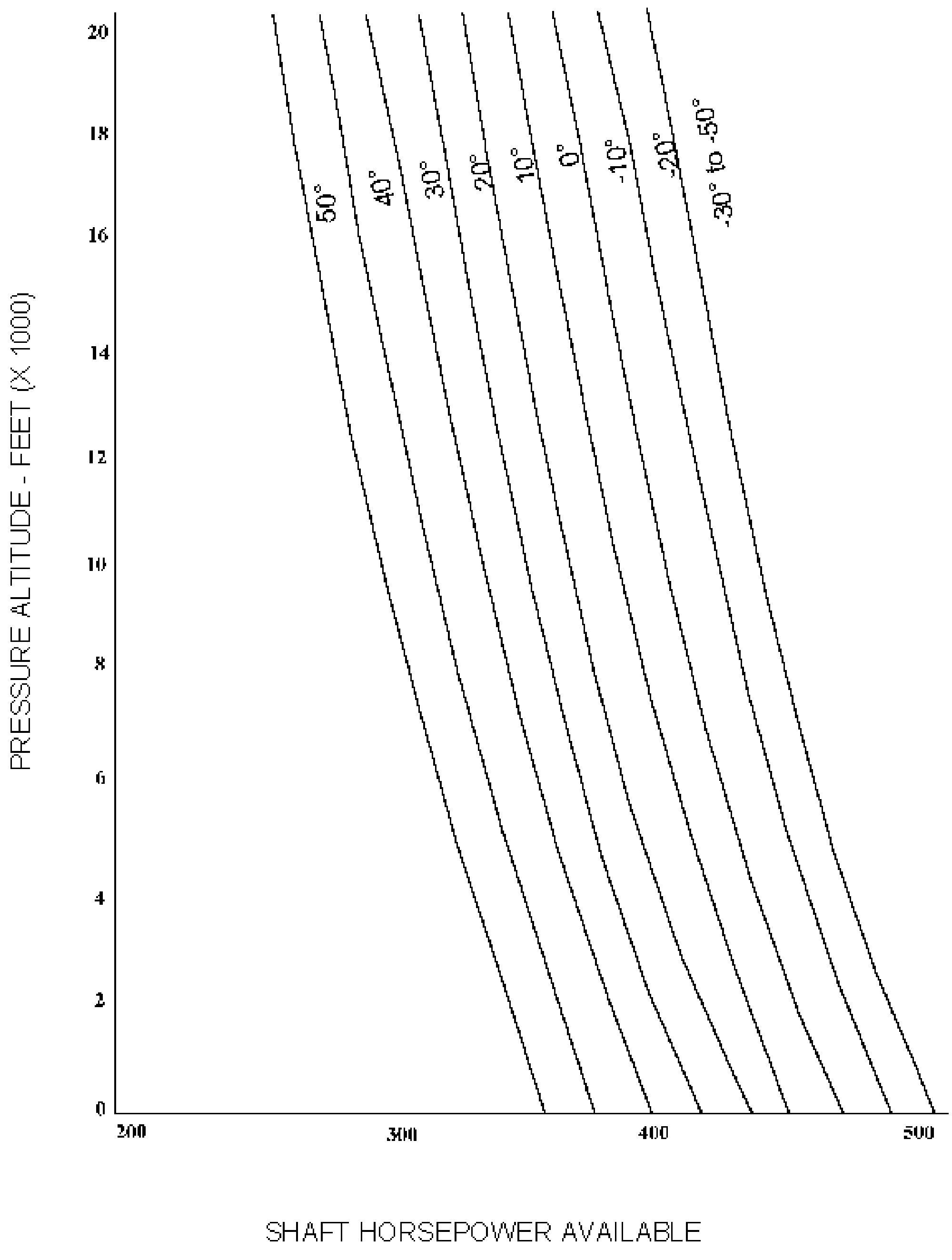


FIGURE AC 27.45-4 UNINSTALLED TAKEOFF POWER AVAILABLE

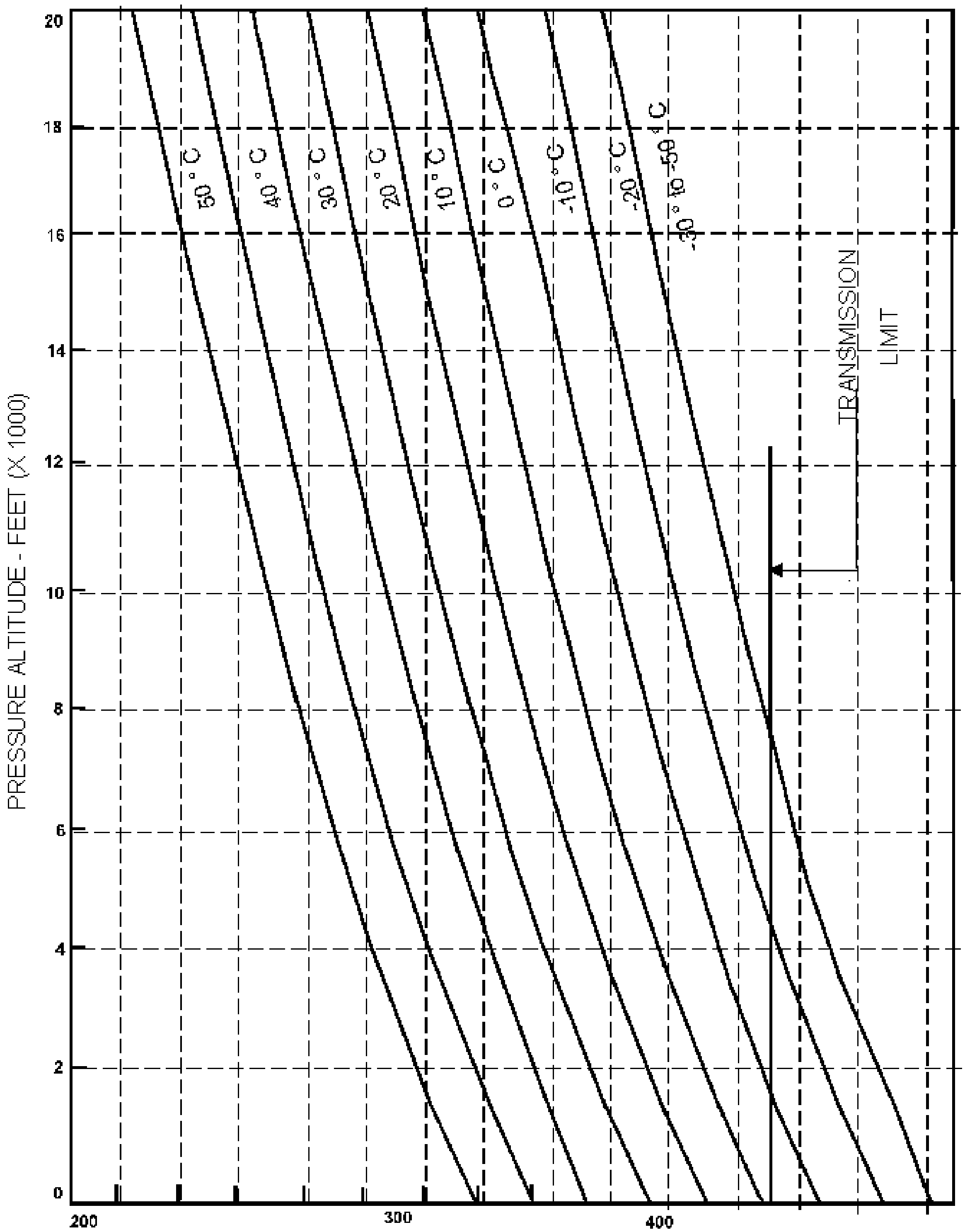


FIGURE AC 27.45-5 INSTALLED TAKEOFF POWER AVAILABLE, ANTI-ICE OFF, 400 RPM

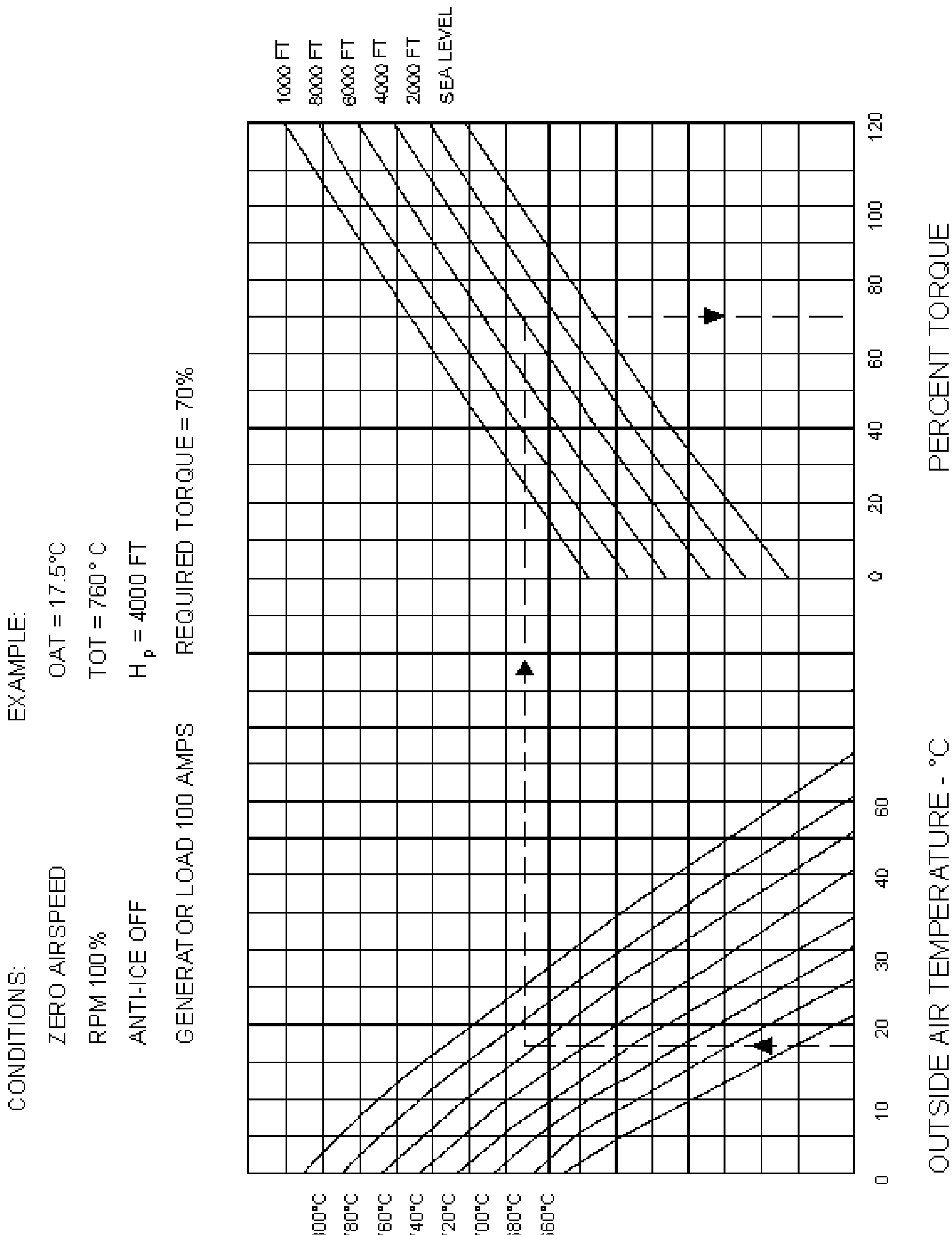


FIGURE AC 27.45-6 POWER ASSURANCE CHECK CHART

AC 27.51. <u>§ 27.51 TAKEOFF</u>.

0 a. <u>Explanation</u>. Section 27.51 details the conditions under which takeoff data must be obtained. The flight manual must contain the technique(s) to be used to obtain the published flight manual takeoff procedures. Technique should not be confused with exceptional pilot skill and/or alertness as mentioned in § 27.51. Because rotorcraft differ, different pilot techniques are sometimes required to achieve the safest and most optimum takeoff performance. The recommended technique that is published in the flight manual must be determined to be one that the operational pilot can duplicate using the minimum amount of type design cockpit instrumentation and the minimum crew. Only rotorcraft takeoff techniques will be covered in this section.

b. <u>Background</u>.

(1) Certain special takeoff techniques are necessary when a rotorcraft is unable to take off vertically because of altitude, weight, power effects, or operational limitations. The recommended technique used to take off under such conditions is to accelerate the rotorcraft in-ground-effect (IGE) to a predetermined airspeed prior to climbout. Takeoff tests are performed to determine the best repeatable technique(s) for a particular rotorcraft over the range of weight and altitude for which certification is requested.

(2) Utilizing the total power available to execute a takeoff may not be operationally feasible due to such items as HV or aircraft attitude constraints. In such situations, hover power required plus some power increment may be the maximum recommended for use.

(3) Wheel or skid height should be not less than that demonstrated satisfactorily for the high speed, low altitude portion of the HV curve, or that height below which ground contact may occur when accomplishing takeoff procedures.

(4) For rotorcraft fitted with wheels, a running takeoff procedure may be accepted.

c. <u>Procedures</u>.

(1) There are different takeoff profiles which may be used to complete a maximum performance takeoff in a rotorcraft. The manufacturer will normally determine which method is best for a particular rotorcraft. The most commonly accepted method is the hover and level acceleration technique. In this technique, the rotorcraft is stabilized in a hover at the reference height. From the stabilized hover, the rotorcraft is accelerated to the climbout airspeed using the predetermined takeoff power. When the desired climbout airspeed is achieved, the rotorcraft is rotated and the climbout is accomplished at the scheduled airspeed(s) and constant rotor RPM. Power adjustments may be accomplished to maintain the targeted power except where procedure requires high workload outside the cockpit (i.e., that portion of takeoff where

horizontal acceleration close to the ground has pilot scan outside the cockpit and adjustment of engine torque or temperature would require an undue increase in workload). The recommended takeoff procedure must be demonstrated to remain clear of the HV "avoid" areas without requiring exceptional piloting skill or exceptionally favorable conditions.

(2) The hover reference height is established as the minimum skid or wheel height above the takeoff surface from which a takeoff can consistently be accomplished in zero wind without contacting the runway surface. The takeoff must be accomplished with power fixed at the power required to hover at the hover reference height and must not require exceptional piloting skill to avoid runway surface contact.

AC 27.65. <u>§ 27.65 (Amendment 27-14) CLIMB: ALL ENGINES OPERATING</u>.

a. <u>Explanation</u>.

(1) Rotorcraft other than helicopters.

(i) Section 27.65 requires that the steady rate of climb be determined for each rotorcraft other than helicopter with maximum continuous power on each engine for the range of weights, altitudes, and temperatures for which certification is requested. Equivalent levels of safety have been found wherein the applicant was allowed to select a climb airspeed that was not the actual V_Y. The selected airspeed must be consistent with the speed used to show compliance with such items as cooling, stability, etc. The rate of climb resulting from the selected climb airspeed versus that from the actual V_Y shall not differ to an extent that a pilot will be encouraged, by appreciable increases in climb performance, to fly a climb airspeed different from that published in the flight manual.

(ii) For rotorcraft other than helicopters, the climb performance data obtained above must be used to show that a minimum climb gradient can be achieved for each weight, altitude, and temperature within the range for which certification is required. This gradient must be at least 1:10 if testing is done to determine the required takeoff distance over a 50-foot obstacle. If this option is selected, an explanation of the takeoff distance determination requirements and procedures may be found in paragraph AC 29.63 of AC 29-2C.

(iii) If takeoff distance is not determined, the minimum climb gradient must be 1:6 for standard sea level conditions.

(2) For helicopters, V_Y must be determined for standard sea level conditions at maximum weight using maximum continuous power on each engine. Although not required, the steady rate of climb may be determined using the procedure in paragraph AC 27.65c of this section (Procedure to Determine All-Engine-Operating Climb Performance).

(3) For helicopters, if V_{NE} at any altitude is less than the maximum gross weight sea level standard day condition V_Y, the steady rate of climb must be determined at the climb speed(s) selected by the applicant not to exceed V_{NE}. The climb performance must be determined from 2,000 feet below the altitude from where V_{NE} intersects V_Y up to the maximum altitude for which certification is requested. This should be done utilizing maximum continuous power on each engine with the landing gear retracted.

b. Procedure to Determine V_Y.

(1) Sawtooth climbs may be used to determine V_Y. If such a technique is used, climbs should be flown in pairs on opposite headings 90° to the wind at the test altitude. This procedure will minimize any windshear effects. All testing must be done in smooth air. Windshear is usually an indication of unstable air or a temperature inversion and must also be avoided. The climbs are flown on reciprocal headings for approximately 5 minutes or through an altitude band using maximum continuous power at a constant airspeed. Periodic power adjustments may be necessary. Additional reciprocal heading climbs must also be conducted at different airspeeds above and below the airspeed at the lowest point of the power required versus airspeed curve. This technique can be repeated at different altitudes to obtain V_Y throughout the altitude range.

(2) Level flight performance (speed power) may also be used to determine V_Y. The testing should be done in smooth air. The advantage of this method is that less time is required, and the accuracy is equivalent to the sawtooth climb method. The test can be repeated at various altitudes to determine the V_Y throughout the altitude range desired for the rotorcraft. The test at each altitude should be conducted at a constant weight over sigma (W/σ). The test is normally started at the desired W/σ with maximum continuous power, or at V_{NE}, in level flight. A series of points should be taken, reducing airspeed 10 to 15 knots between points, with the lowest speed point around 20 to 30 knots. Weight should be computed for each point and the test altitude adjusted to maintain a constant W/σ. After the data are reduced to standard day conditions, the minimum power required airspeed will be the V_Y speed.

(3) Prior to the flight test, the rotorcraft should be ballasted to the desired gross weight and the critical center of gravity. The airspeed should be stabilized prior to data acquisition. Data to be recorded includes time, altitude, airspeed, ambient temperature, engine parameters, torque(s), rotor RPM, fuel reading, aircraft heading, external configuration, etc. Power setting, weight, and climb airspeed should be planned prior to flight. For some turboshaft engines, temperature and/or engine speed limits may be reached prior to a limiting torque. The test team should verify that the resulting power utilized in these tests closely approximates the power producing capabilities of a minimum installed specification engine.

c. Procedure to Determine All-Engine-Operating Climb Performance.

(1) Background. Continuous climbs are conducted at the appropriate climb airspeeds as outlined above in order to validate the rotorcraft's climb performance. By-products are a qualitative evaluation of the rotorcraft handling characteristics in a climb and engine data to assist in the determination of installed power available.

(2) Techniques. The climbs are conducted on reciprocal headings at the established airspeed(s) through the target altitude range. The same parameters are recorded as during sawtooth climbs. The rotorcraft will usually climb very rapidly during the first few thousand feet; therefore, the data acquisition method must be timely if accurate results are expected. This procedure is usually repeated at weight extremes. The resulting data must then be corrected for power and weight. Power and weight corrections are satisfactory, provided the test powers and weights closely approximate the target values to make the weight and power corrections small. Once this data is finalized and corrected for all the flight test variables, interpolation for intermediate weights can be made with a high degree of reliability. If the rotorcraft has any stability augmentation system, vent systems, etc., which may influence the climb performance, then it must be accounted for. Caution should be taken that anti-ice, air-conditioning, etc., are not on unless the performance is being established specifically for those conditions.

AC 27.65A. § 27.65 (Amendment 27-33) CLIMB: ALL ENGINES OPERATING.

a. Explanation. Amendment 27-33 added the requirement to determine the steady rate of climb, for helicopters, from sea level up to the maximum altitude for which certification is requested. Although not specifically stated in the rule, the rate of climb should be determined at V_Y or, if V_{NE} at any altitude is less than the maximum gross weight sea level standard day condition V_Y, the steady rate of climb at these altitudes must be determined at a climb speed(s) selected by the applicant not to exceed V_{NE}.

b. Procedures. The policy material pertaining to the procedures outlined in this section remain in effect.

AC 27.67. § 27.67 (Amendment 27-23) CLIMB: ONE ENGINE INOPERATIVE.

a. Explanation.

(1) Section 27.67 requires that for multiengine normal category rotorcraft, the steady rate of climb or descent with one engine inoperative must be determined at V_Y (or at the speed for minimum rate of descent) for maximum gross weight.

(2) The rate of climb (or descent) will be determined with the critical engine inoperative and the remaining engine(s) at maximum continuous or 30-minute minimum specification installed power available values. The landing gear should be retracted if it is retractable.

b. Procedures.

(1) The procedure discussed in paragraph AC 27.65 for all-engines-operating climb performance is also applicable to the OEI condition. For twin-engine rotorcraft that are shown not to have a “critical engine” with respect to performance characteristics, both engines may be used to simulate the appropriate single-engine power available during these tests.

(2) Adequate testing must be accomplished to determine the rotorcraft’s OEI climb performance at maximum gross weight for all variations in altitude and temperature for inclusion in the Rotorcraft Flight Manual.

AC 27.71. § 27.71 (Amendment 27-21) GLIDE PERFORMANCE.

a. Explanation.

(1) Performance capabilities during stabilized autorotative descent are useful tools to assist the pilot when all engines fail. This information is also useful in determining the suitability of available landing areas along a given route segment.

(2) Two speeds are of particular importance, the speed for minimum rate of descent and the speed for best angle of glide. These speeds along with glide distance information are required as flight manual entries per § 27.1587. The speed for minimum rate of descent is useful for engine failure conditions at higher altitudes and the pilot is required to perform some time-related task, engine restart, float inflation, radio calls, etc. The speed for best angle of glide is a somewhat higher speed that is of particular use when it is necessary to reach a distant landing area. These speeds, when utilized in conjunction with appropriate rotor RPM and glide angle (or rate of descent) can be used to calculate the maximum horizontal distance available from a particular altitude assuming zero wind conditions.

(3) A third speed, recommended autorotation speed, may be provided in addition to minimum rate of descent speed and maximum glide angle speed. The recommended speed for autorotation is usually optimized to assure an effective flare capability and yet be slow enough to allow a controlled, relatively slow touchdown condition. Recommended autorotation speed is ordinarily between the minimum rate of descent and maximum glide angle speeds. The recommended autorotation speed may be provided in the Rotorcraft Flight Manual. The relationship between minimum rate of descent, best glide angle, and recommended autorotation speed is shown in figure AC 27.71-1.

(4) Forward center of gravity is usually critical; however, center of gravity effects should be spot-checked to confirm this for a given design.

b. Procedures.

(1) Tests are conducted at speeds which bracket the anticipated speeds for minimum rate of descent and best glide angle. On a power required plot, the speed for minimum power required approximates the speed for minimum rate of descent. The speed for maximum range glide may be estimated by drawing a tangent from the origin to the power required curve.

(2) Autorotative performance tests may be conducted in conjunction with the climb performance tests. The required data are similar for both tests and it is sometimes convenient and efficient to run alternating climbs and descents through a desired altitude band. Descents should be conducted on reciprocal headings and results averaged in the same manner as climb performance tests.

(3) A reduction in rotor RPM from the normal power-on value may enhance autorotative performance. If the applicant wishes to develop autorotative performance at RPM values significantly below the governing or power-on range, the practicality of reducing and controlling RPM at the lower value and of then increasing RPM as a landing is approached, must be considered. At low weights and low density altitudes, full down collective may automatically produce lower RPM values and this condition is, of course, acceptable provided the approved power-off RPM range is not exceeded.

(4) During autorotation tests, care must be taken to make certain that no engine power is delivered to the rotor drive system since a very small amount of power can have a large effect on descent performance.

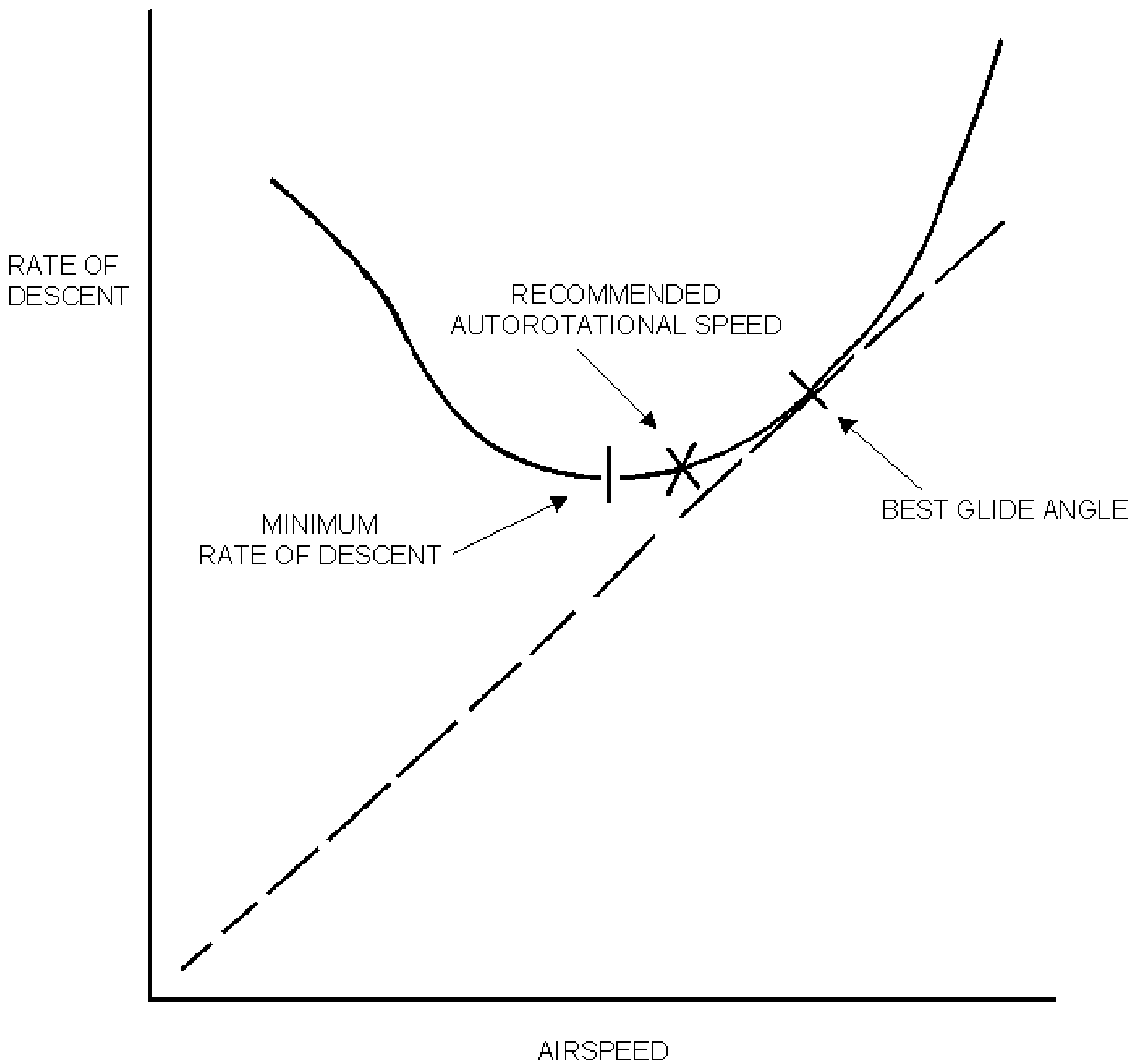


FIGURE AC 27.71-1 AUTOROTATIONAL CHARACTERISTICS - TYPICAL

AC 27.73. § 27.73 PERFORMANCE AT MINIMUM OPERATING SPEED.

a. Explanation.

(1) The word "hover" applies to a rotorcraft that is airborne at a given altitude over a fixed geographical point regardless of wind. Pure hover is accomplished only in still air. For the purpose of this manual, the word "hover" will mean pure hover.

(2) The regulatory requirement for hover performance, § 27.73, refers to hover in ground effect (IGE). For some applications, such as external load operations, hover performance out-of-ground effect (OGE) is necessary; however, it is not required by this section. Hover OGE is that condition, where an increase in height above the ground will not require additional power to hover. Hover OGE is the absence of measurable ground effect. It can be less than one rotor diameter at low gross weight increasing significantly at high gross weight. The lowest OGE hover height at gross weight may be approximated by placing the lowest part of the vehicle one and one-half rotor diameters above the surface.

(3) The objective of hover performance tests is to determine the power required to hover at different gross weights, ambient temperatures, and pressure altitudes. Using nondimensional power coefficients (C_p) and thrust coefficients (C_t) for normalizing and presenting test results minimizes the amount of data required to cover the rotorcraft's operating envelope.

(4) Hover performance tests must be conducted over a sufficient range of pressure altitudes and weights to cover the approved ranges of those variables for takeoff and landing. Additional data should be acquired during cold ambient temperatures, especially at high altitudes, to account for possible Mach effects.

(5) The hover ceiling for which data should be obtained and subsequently presented in the flight manual should be the same height consistent with the minimum hover height demonstrated during the takeoff tests. Refer to paragraph AC 27.51 for the procedure to determine this hover height.

b. Procedures.

(1) Two methods of acquiring hover performance data are the tethered and the free flight techniques. The tethered technique is accomplished by tethering the rotorcraft to the ground using a cable and load cell. The load cell and cable are attached to the ground tie-down and to the rotorcraft cargo hook. The load cell is used to measure the rotorcraft's pull on the cable. Hover heights are based on skid or wheel height above the ground. During tethered hover tests, the rotorcraft should be at light gross weight. The rotorcraft will be stabilized at a fixed power setting and rotor speed at the appropriate skid or wheel height. Once the required data are obtained, power should be varied from the minimum to the maximum allowed at various rotor RPM. This technique will produce a large C_t/C_p spread. The load cell reading is recorded for each

stabilized point. The total thrust the rotor produces is equal to the rotorcraft's gross weight plus the weight of the cables and load cell plus cable tension. Care must be taken that the cable tension does not exceed the cargo hook limit or load capacity of the tie-down. For some rotorcraft, it may be necessary to ballast the rotorcraft to a heavy weight in order to record high power hover data.

(2) The pilot maintains the rotorcraft in position so that the cables and load cell are perpendicular to the ground. To ensure the cable is vertical, two outside observers, one forward of the rotorcraft and one to one side, can be used. Either hand signals or radio can be used to direct the pilot. The observers should be provided with protective equipment. Positioning can also be accomplished by attaching two accelerometers to the load cell which sense angle or movement along the longitudinal and lateral axes. Any displacement of the load cell will be reflected on instrumentation in the cockpit, and by reference to this instrumentation, the rotorcraft can be maintained in the correct position. Increased caution should be utilized as tethered hover heights are decreased because the rotorcraft may become more difficult to control precisely. The tethered hover technique is especially useful for OGE hover performance data because the rotorcraft's internal weight is low and the cable and load cell can be jettisoned in the event of an engine failure or other emergency.

(3) To obtain consistent data, the wind velocity should be less than 3 knots as there are no accurate methods of correcting hover data for wind effects. Rotorcraft with high downwash velocities may tolerate higher wind velocities. The parameters usually recorded at each stabilized condition are:

(i) Engine torque.

(ii) Rotor speed.

(iii) Ambient temperatures.

(iv) Pressure altitude.

(v) Fuel used (or remaining).

(vi) Load cell reading.

(vii) Generator(s) load.

(viii) Wind speed and direction.

As a technique, it is recommended the rotorcraft be loaded to a center of gravity near the hook to minimize fuselage angle changes with varying powers. All tethered hover data should be verified by a limited spotcheck using the free flight technique. The free flight technique as contained in paragraph AC 27.73b(4) will determine if any problems, such as load cell malfunctions, have occurred. The free flight hover data must fall within the allowable scatter of the tethered data.

(4) If there are no provisions or equipment to conduct tethered hover tests, the free flight technique is also a valid method. The disadvantage of this technique as the primary source of data acquisition is that it is very time consuming. In addition a certain element of safety is lost OGE in the event of an emergency. The rotorcraft must be reballasted to different weights to allow the maximum C_t/C_p spread. When using the free flight technique, either as a primary data source or to substantiate the tethered technique, the same considerations for wind, recorded parameters, etc., as used in the tethered technique apply. Free flight hover tests should be conducted at CG extremes to verify any CG effects. If the rotorcraft has any stability augmentation system which may influence hover performance, it must be accounted for.

(5) It is extremely difficult to determine when a rotorcraft is hovering OGE at high altitudes above ground level since there is no ground reference. In a true hover, the rotorcraft will drift with the wind. Numerous techniques have been tried to allow OGE hover data acquisition at high altitudes, all of which have resulted in much data scatter. Until a method is proposed and found acceptable to the FAA/AUTHORITY, OGE hover data must be obtained at the various altitude sites where IGE hover data are obtained. Hover performance can usually be extrapolated up to a maximum of 4,000 feet.

AC 27.75. § 27.75 (Amendment 27-14) LANDING.

a. Explanation.

(1) This rule incorporates all of the landing requirements for Part 27 rotorcraft.

(2) As with other flight maneuvers, landings must be accomplished with acceptable flight and ground characteristics using normal pilot skills. Reasonable sampling and extrapolation methods are, of course, allowed. General guidance on those subjects is given in paragraph AC 27.45. As in other performance areas, engines must be operated within approved limits.

(3) Landing. Approach and landing path requirements are stated in general terms in paragraphs (a)(1) and (a)(2) of § 27.75. The approach path must allow smooth transition for a one-engine-inoperative landing and adequate clearance from potentially hazardous HV combinations.

(4) All-engine-out landing. Section 27.75(b) contains the certification requirement for “last” engine failure and all-engines-inoperative landing. The rule states that it must be possible to make a safe landing after complete power failure during normal cruise. It is not intended that all engines be failed simultaneously, although complete power failure has occurred in twin-engine rotorcraft with Category A engine isolation. This requirement assures that in the event of cockpit mismanagement, fuel

exhaustion, improper maintenance, fuel contamination, or unforeseen mechanical failures, a safe autorotation entry can be made and a safe power-off landing can be affected. Two separate aspects of this rule are normally evaluated at different times during the test program. The “last” engine failure is normally evaluated during cruise or V_{NE} engine failure testing where instrumentation and critical loading have been established for those test conditions. The all-engine-out landing is ordinarily conducted in conjunction with an HV or landing distance phase where ground instrumentation and safety equipment are available.

b. Procedures.

(1) Instrumentation/Equipment. Aircraft instrumentation may include engine and flight parameters, control positions, power lever position, and landing gear loads. A record of rotor RPM at touchdown is necessary to assure it does not exceed transient limits. Rotor RPM at touchdown may be lower than the minimum transient limit for flight, provided stress limits are not exceeded. A crash recovery team with the support of a fire engine is highly desirable.

(2) The one-engine-inoperative landing is similar in many respects to the HV tests described in paragraph AC 27.79. Most of the comments, cautions, and techniques for HV also apply here even though the typical flight conditions are less critical than limiting HV points due to a lower power level and an established rate of descent. The approach is made at a predetermined speed with one engine inoperative. The speed is reduced and the rotorcraft is flared to a conventional one-engine-inoperative landing.

(3) Power. Power should be limited to minimum specification values on the operating engine(s). This may be accomplished by adjustment of engine topping to minimum specification values for the range of atmospheric variables to be approved. This is frequently done by installing an adjustable device in the throttle linkage with a control in the cockpit so that engine topping can be accurately adjusted for varying ambient conditions. With such a device in the control system it becomes vitally important to check topping power prior to each test sequence.

(4) Aircraft Loading. Aft center of gravity is usually most critical because visibility constraints limit the degree to which the pilot can see the landing surface during the flare. If a weight effect is shown, a minimum of two weights should be flown at each test altitude. One weight should be the maximum weight for prevailing conditions, and the other should provide a sufficient spread to validate weight accountability.

(5) All-engine-out landing.

(i) Several procedures can be utilized to demonstrate compliance with the all-engine-out landing requirement. As discussed in the explanation portion of this paragraph, § 27.75(b) contains two separate requirements. One is the ability to transition safely into autorotation after failure of the last operative engine. The second aspect of this rule requires that a landing from autorotation be possible. The second requirement is discussed below. The maneuver is entered by smoothly reducing power at an optimum autorotation airspeed at a safe height above the landing surface. If a complete company test program has documented an all-engine-out landing to the GW/ (gross weight/density ratio) limit, verification tests may be initiated at those limiting weight conditions. If not, buildup testing should be initiated at light weight. This test is ordinarily conducted at mid center of gravity. Typically, all altitudes may be approved with two weight limit landings-one at sea level and one near maximum takeoff and landing altitude.

(ii) Demonstrated compliance with this requirement is intended to show that an autorotative descent rate can be arrested, and forward speed at touchdown can be controlled to a reasonable value (less than 40 KTAS is recommended) to ensure a reasonable chance of survivability for the all engine failure condition. On multiengine rotorcraft, rotor inertia is typically lower than for single-engine rotorcraft. RPM decays rapidly when the last engine is made inoperative. Due to this relatively low inertia level, considerable collective may be needed to prevent rotor overspeed conditions when the rotorcraft is flared for landing. Also, when testing the final maximum weight points, the pilot should anticipate a need for considerable collective pitch to control rotor overspeed during autorotative descent, particularly at high altitude WAT limiting conditions. Some designs incorporate features which may lead to rotorcraft damage in testing this requirement (e.g., droop stop breakage or loss of directional control with skids) if landings are conducted to a full stop with the engines cut off.

(iii) The intent of this rule is to demonstrate controlled touchdown conditions and freedom from loss of control or apparent hazard to occupants when landing with all engines failed. In these cases compliance can be demonstrated by leaving throttles in the idle position and ensuring no power is delivered to the drive train. Also, computer analysis may be used in conjunction with simulated in-flight checks to give reasonable assurance that an actual safe touchdown can be accomplished. Another method may be to make a power recovery after flare effectiveness of the rotorcraft has been determined. Other methods may be considered if they lead to reasonable assurance that descent can be arrested and forward speed controlled to allow safe landing with no injury to occupants when landing on a prepared surface with all engines failed. Regardless of the method(s) used to comply with this requirement, careful planning and analyses are very important due to the potentially hazardous aspects of power off simulation and landing of a multiengine rotorcraft totally without power. The all-engine-inoperative landing test is ordinarily done in conjunction with height velocity tests because ground and onboard instrumentation requirements are the same for both tests.

(6) Prior to conducting these tests, the crew should be familiar with the engine inoperative landing characteristics of the rotorcraft. The flight profile may be entered in the same manner as a straight-in practice autorotation. It is recommended that for safety reasons idle power be used if a “needle split” (no engine power to the rotor) can

be achieved. In some cases, a low engine idle adjustment has been set to assure needle split is attained. In other cases a temporary detent between idle and cutoff was used on the throttle. In a third case the engine was actually shut down on sample runs to verify that the engine power being delivered was not materially influencing landing capability or landing distances. The flare is maintained as long as is reasonable to dissipate speed and build RPM Rotor RPM must stay within allowable limits. Aft center of gravity is ordinarily critical due to visibility and flarability. Following the flare, the rotorcraft is allowed to touch down in a landing attitude. Rotor RPM at touchdown should be recorded, and it must be within allowable structural limits.

AC 27.79. § 27.79 (Amendment 27-14) LIMITING HEIGHT-SPEED ENVELOPE.

a. Explanation.

(1) The height-speed envelope is normally referred to as the height-velocity (HV) diagram. It defines an envelope of airspeed and height above the ground from which a safe power-off or OEI landing cannot be made. The diagram normally consists of three portions: (a) the level flight (cruise) portion, (b) the takeoff portion, and (c) the high speed portion. See figure AC 27.79-1. The high speed portion is omitted on occasions when it can be shown that the rotorcraft can suffer an engine failure at low altitude and high speed (up to V_H) and make a successful landing or climb out on the remaining engine(s).

(2) Power failure, engine failure, throttle chop, or other similar terms used in this discussion mean a simulated engine failure. The actual shutdown of an engine to simulate an engine failure should not be necessary if the simulated procedure ensures that the engine power is suddenly removed from driving the rotor and remains so. The normal fuel control deceleration schedule is usually satisfactory for the power removal for turbine engines but the flight/ground idle speed may have to be set lower than normal for HV testing.

(3) The avoid areas of the HV diagram are separated by the takeoff corridor. This corridor should be wide enough to consistently permit a takeoff flight path clear of the HV diagram using normal pilot skill. The takeoff corridor should always permit a minimum of ±5 knots clearance from critical portions of the diagram.

(4) The knee of the curve separates the takeoff portion from the cruise portion and is defined as the highest speed point on the low speed portion of the HV envelope. Altitudes above this point are considered cruise, or “fly-in,” points, and these test points require a minimum time delay of 1 second between throttle chop and control actuation (reference § 27.143(d)). Altitudes below the knee represent takeoff profile points. For test points in the takeoff portion, takeoff power (or a lower power selected by the applicant as an operating procedure) and normal pilot reaction time for corrective control actuation will be used.

(5) Since the HV diagram may represent the limiting capabilities of the rotorcraft, each test point should be approached with caution. The manufacturer's buildup program should be reviewed to determine the amount of conservatism in the HV diagram (if any). It should be remembered that the operational pilot will be operating at or near the HV diagram without the benefit of a buildup program. Buildup testing is necessary, and it is most important to vary only one parameter at a time to prevent surprises. Light weight testing is ordinarily conducted first. High and low hover points are approached from above and below respectively. Portions near the knee are initially evaluated at high speed with subsequent backing down of the speed. In most rotorcraft the effective flare airspeed is critical. At airspeeds slightly below this value, the ability to arrest and control descent rates through use of an aft cyclic flare may be greatly diminished. Extreme care should be exercised when "backing down" to lower speeds.

(6) In addition to the on-board and ground instrumentation, a motion picture camera or other position measuring equipment should cover each run.

(7) For FAA/AUTHORITY tests, the minimum required crew and the minimum instrument panel display presented for certification should be used. Ground safety equipment should be provided.

(8) This test is the least predictable of all the performance items. Therefore, the expansion and extrapolation of test data are questionable. Weight may not be extrapolated to higher values. In order to extrapolate HV data to higher altitudes, any analytical method must have FAA/AUTHORITY approval. In lieu of pure analytical methods, simulations have been used successfully, especially for multiengine rotorcraft. In either case, the maximum allowable extrapolation should be limited to 2,000 feet density altitude (H_D). HV test weights for normal category rotorcraft are the maximum weight at sea level and some lessor weight at high density altitudes. The high density altitude HV curve needs to be defined only to 7,000 feet and may be a lower altitude if the rotorcraft does not have the performance capabilities to attain 7,000 feet. A weight less than the maximum weight may be used to define the high density altitude HV curve, but this weight should not be less than the maximum weight that will allow hovering out-of-ground effect. For a given diagram, typical weight reductions that are necessary as altitude is increased can be conservatively estimated by maintaining a constant gross weight divided by density ratio, GW/σ. See figure AC 27.79-2, Part A. If weight is not varied, an enlarged HV diagram is required for safe power-off landing as density altitude is increased. See figure AC 27.79-2, Part B. Another method of presentation is to show varying weights at a constant density altitude. (See figure AC 27.79-2, Part C.)

(9) Vertical takeoff and landing (VTOL) testing normally does not require separate HV testing. The takeoff and landing tests take on the combined characteristics of takeoff, landing, and HV tests.

b. <u>Procedures</u>.

(1) Instrumentation.

(i) Ground Station. The ground station must have equipment and instrumentation to determine wind direction and velocity, outside air temperature, and if the test rotorcraft has reciprocating engines, humidity. Since the tests must be conducted in winds of 2 knots or less, a smoke generator is highly recommended to show both flightcrew and ground crew personnel the wind direction and velocity at any given time. Additionally, the location of the ground station should be such that it is free of rotor downwash at all times. Motion picture or phototheodolite and radio equipment will be necessary to properly conduct the test program. The use of telemetry equipment is desirable if the location of the test site and the magnitude of the test program make it practical.

(ii) Airborne Equipment (Test Rotorcraft). Necessary installed test equipment may include photopanels and/or recorders for recording engine parameters, control positions, landing gear loads, landing gear deflections, airspeed, altitude, and other variables. An external light attached to the rotorcraft (or any other means of identifying the engine failure point to the ground camera or phototheodolite) is needed to identify the exact time of engine failure and may also be used to synchronize the ground recorder with the airborne recorded data.

(2) Analytical Prediction. The HV diagram can be estimated by analytical means and this is recommended prior to test. HV, however, is the least predictable of all rotorcraft performance and because of this, the expansion and extrapolation of test data must be done with great care. Test weight may not be extrapolated. All test points should be approached conservatively with some speed or altitude margin. If the applicant has conducted a comprehensive HV flight test program to validate his analytical predictions, much preliminary testing can be eliminated. In any case, the maximum allowable extrapolation from flight test conditions is 2,000 feet density altitude, and an approved analytical and/or simulation method must be utilized for extrapolation.

(3) Power.

(i) The appropriate power level before engine failure for the low and high hover points is simply the power required to hover at the prevailing hover conditions. The appropriate power condition prior to failure of the engine for points below the knee is takeoff power or a lower value if approved as an operating procedure. For cruise or “fly-in” points above the knee, the appropriate condition is power required for level flight.

(ii) The applicable power failure conditions are listed in § 29.79(b). Power should be completely cut for normal category rotorcraft. For multiengine rotorcraft with Category A engine isolation, only one engine need be failed and the desired topping power (for the remaining engine(s)) should be set prior to the test. This power value will need adjustment as ambient conditions change. The power can be takeoff power (TOP), 2 ½-minute power, or some calculated lower power for simulating hot day or higher density altitude conditions. Power is verified and recorded by the pilot by "topping" the engine(s) prior to engine failure tests. Care must be taken to ensure that this power value is no more than that which would be delivered by a minimum specification engine under the ambient conditions to be approved.

(4) Test Loadings. Weight extrapolation is not permitted for HV. Therefore, the test weight must be closely controlled. Ballast or fuel should be added frequently to maintain the weight within -1 to +5 percent when testing final points. Ordinarily, tests are conducted at a mid center of gravity unless a particular loading is expected to be particularly critical.

(5) Landing Gear Loads.

(i) Instrumented landing gear can be a great help in evaluating test results. This information can be telemetered to a ground station or otherwise recorded and displayed for direct reference following each landing.

(ii) Any landing which results in permanent deformation of aircraft structure or landing gear beyond allowable maintenance limits is considered an unsatisfactory test point.

(6) Piloting Considerations. In verifying the HV diagram, the minimum certificated instrument panel display and minimum crew should be used in order not to mislead the operational pilot who has no test equipment available and may have no copilot to assist. Three distinctly different flight profiles are utilized in developing the diagram.

(i) High Hover. A stabilized out-of-ground-effect (OGE) hover condition prior to power failure is essential. A minimum 1-second time delay between power failure and initial control actuation is utilized. Following the time delay, the primary concern is to quickly lower collective and to gain sufficient airspeed to allow an effective flare approaching touchdown. While the immediate development of airspeed is necessary, the dive angle must be reasonable and must be representative of that expected in service. While initial aircraft attitude will vary between models and with changing conditions, 10°-20° has been previously applied as a maximum allowable nose down pitch attitude. Use of greater attitudes could result in a diagram which is difficult to achieve and unrealistic for operations in service. Initial testing should start relatively high with gradual lowering of height to the final high hover altitude. A stabilized OGE hover condition prior to power failure is essential. If a stabilized high hover condition cannot be achieved prior to the engine cut, then this point should be tested from a minimum level flight speed. This will result in an open-ended HV diagram. A smoke source or balloon on a long cord is highly desirable since the wind can vary significantly from surface observations to typical high hover altitudes. Vertical speed must be very near zero at the throttle chop. Any climb or sink rate can have a significant influence on the success of the test point. Use of a radar altimeter with a cross check to barometric altitude is essential.

(ii) Low Hover. From the low hover position there is no flare capability and little time for collective reaction. No time delay is applied other than normal pilot reaction. For typical designs the collective may not be lowered after power failure. Lowering of the collective is not permitted because it is not a pilot action which could be expected if an engine failed without notice during a hovering condition in service. Initial lowering of collective immediately after power failure can result in a very high, unconservative low hover height that is unrealistic for operational conditions. If, however, a design is such that a 1-second pilot delay after power failure could be achieved without any appreciable descent, a slight lowering of collective could be allowed.

(iii) Takeoff Corridor. Normal pilot reaction is applied when the engine is made inoperative. At low speeds, collective may be lowered quickly to retain RPM and minimize the time between power failure and ground contact. If airspeed is sufficient for an effective flare, the aircraft is flared to reduce airspeed, retain rotor RPM, and control vertical speed prior to touchdown. Considerable surface area may be needed for a sliding or rolling stop.

(iv) Additional Considerations. The “in-between” points utilize similar techniques. The cruise or “fly-in” points are similar to the high hover point although the steep initial pitch attitudes are not needed as altitude is decreased and airspeed is increased along the curve. The low speed points along the takeoff corridor are similar to the low hover point except that the collective may be quickly lowered and some flare capability may be used as the “knee” is approached. The pilot should be proficient in all normal autorotation landings before conducting HV tests in a single-engine rotorcraft.

(7) Ground Support. Motion picture or theodolite coverage and ground safety equipment are necessary. Communication capability among these elements should be provided. Use of a phototheodolite to compare height/speed with cockpit observations is very desirable.

(8) Verifying the HV Diagram.

(i) A sufficient number of test points must be flown to verify the diagram. The key areas are the knee, high altitude hover, low altitude hover, and low altitude high speed flight. Test points with excessive gear loads, exceptional skill requirements, winds above permissible levels, rotor droop below approved minimum transient RPM, damage to the rotorcraft, excessive power, incorrect time delay, etc., cannot be accepted.

(ii) After the HV diagram is defined, it should be ascertained that the corridor permits takeoffs within ±5 knots of the recommended takeoff profile.

(9) Flight Manual. The flight manual should list any procedures which may apply to specific points (e.g., high speed points) and test conditions, such as runway surface, wave height for amphibious tests, marginal areas of controllability or landing gear response, etc. The HV curve should be presented in the RFM using actual altitude above ground level and indicated airspeed.

(10) Night Evaluation. If a rotorcraft is to be certified for night operation, a night evaluation is required. Simulated engine failures should be conducted along the recommended takeoff path. Landings should also be qualitatively evaluated with an engine failed. Engine failures at critical HV conditions are not required. The intent is to show adequate visibility using aircraft and/or runway lights without requiring a duplication of the daytime HV test program.

(11) Water Landings. For amphibious float-equipped rotorcraft, day and night water landings should be conducted under critical loading conditions with an engine failed. Engine failures should be conducted along the recommended takeoff path. Engine failures at critical HV conditions are not required. The intent is to show similarity to test results over land without requiring a duplication of the HV test program.

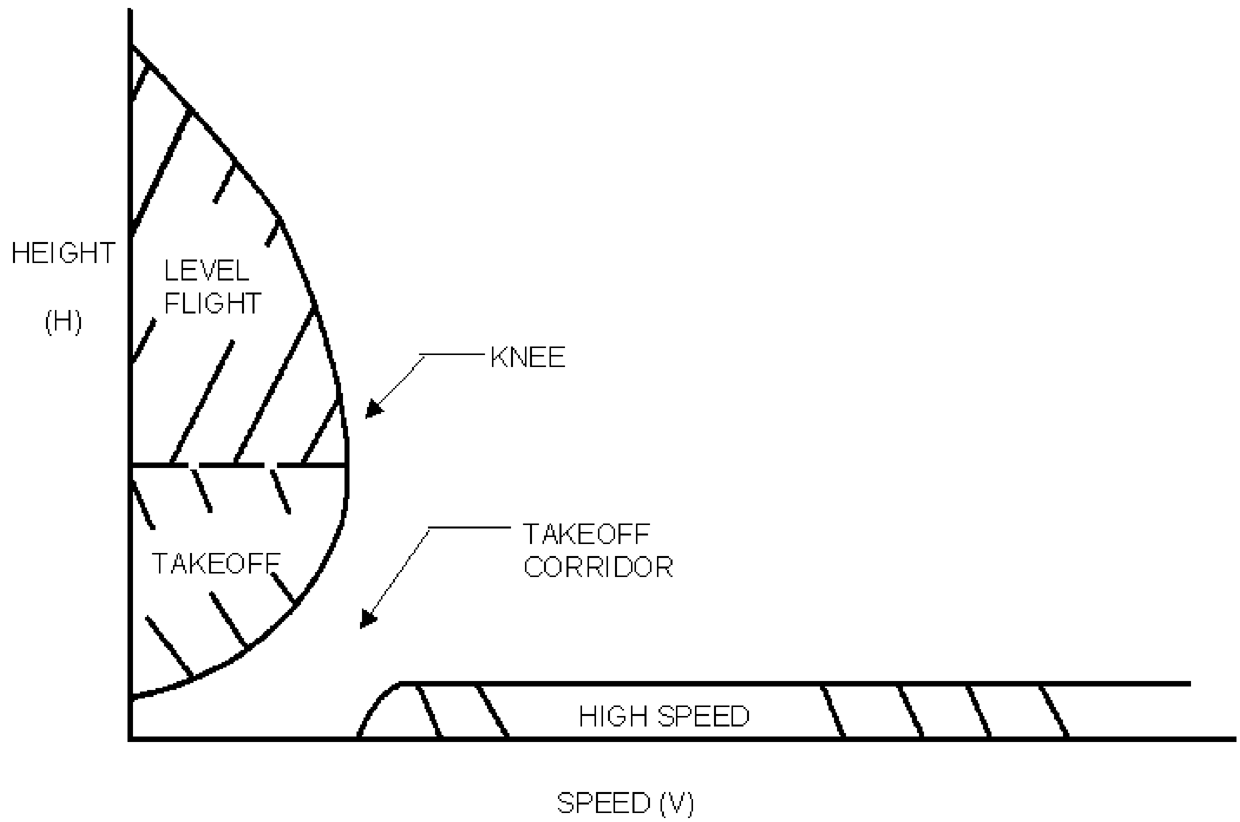


FIGURE AC 27.79-1 HEIGHT - VELOCITY DIAGRAM

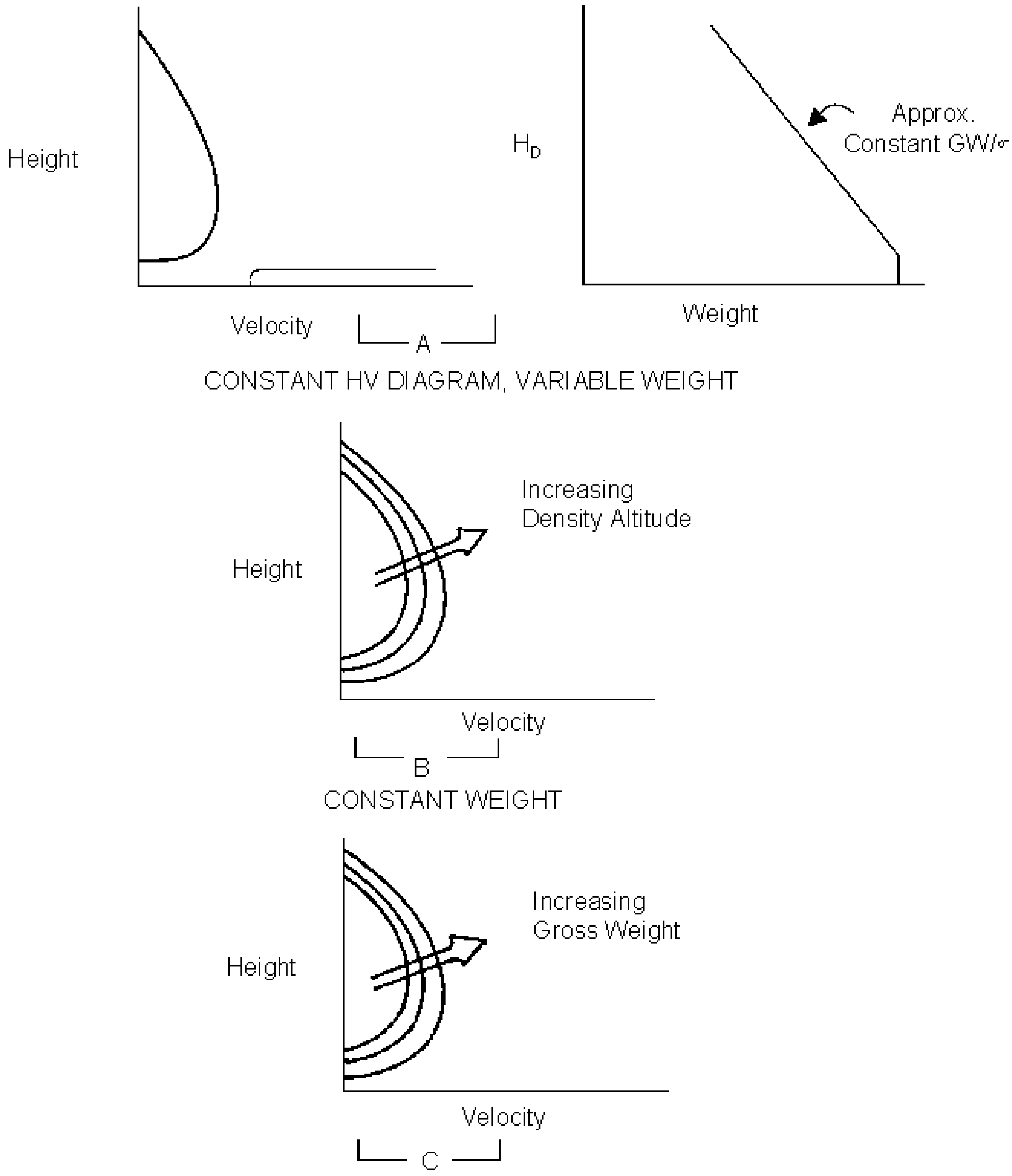


FIGURE AC 27.79-2 ALTITUDE/WEIGHT ACCOUNTABILITY

AC 27.79A. § 27.79 (Amendment 27-21) LIMITING HEIGHT-SPEED ENVELOPE.

a. Explanation. Amendment 27-21 to the regulation redefines the required weight for establishing the HV envelope at altitudes above sea level.

b. Procedures. All of the policy material pertaining to this section remains in effect. In addition, the following applies:

(1) The rotorcraft height-velocity envelope should be established for the maximum gross weight at sea level. At altitudes above sea level, the envelope should be established at not less than the maximum operating weight or OGE hover weight, whichever is lower. If a weight below the OGE hover weight is selected, by definition, that selected weight becomes the maximum operating weight for the rotorcraft at that altitude.

(2) If the HV envelope is established for a maximum altitude less than 7,000 feet, by definition, the maximum takeoff and landing altitude for the rotorcraft may be no higher than that maximum HV altitude. Hover performance information should not be presented for altitudes above the maximum altitude for which the HV envelope is established.

SUBPART B - FLIGHT

FLIGHT CHARACTERISTICS

AC 27.141. § 27.141 (Amendment 27-21) FLIGHT CHARACTERISTICS--GENERAL.

a. Explanation.

(1) This section prescribes the general flight characteristics required for certification of a normal category rotorcraft. Specifically, it states that the rotorcraft shall comply with the flight characteristics requirements at all approved operating altitudes, gross weights, center of gravity locations, airspeeds, power, and rotor speed conditions for which certification is requested. The reference to altitude in § 27.141(a)(1) refers to density altitude. Density altitude is, of course, a function of pressure altitude and ambient temperature, hence the need to account for ambient temperature effects. Additional flight characteristics required for instrument flight are contained in AC 27 Appendix B of this advisory circular.

(2) Generally, the aircraft structural (load level) survey accounts for takeoff power values at speeds up to and including V_Y. At speeds above V_Y, maximum continuous power is assumed. Stress to rotating components usually increases with airspeed and power. If the takeoff power rating exceeds the maximum continuous power rating, and the structural survey has been conducted under the assumption that takeoff power is not used at speeds above V_Y, the Rotorcraft Flight Manual must limit takeoff power to speeds of V_Y and below. If takeoff power is structurally substantiated throughout the flight envelope, and appropriate portions of the controllability, maneuverability, and trim requirements of §§ 27.141 through 27.161 are met at takeoff power levels, no flight manual entry is needed. Obviously if transmission limits for maximum continuous (MC) and takeoff power coincide, no special action is needed.

(3) During the flight characteristics testing, the controls must be rigged in accordance with the approved rigging instructions and tolerances. The control system rigging must be known prior to testing. In addition to the normal rigging procedures, any programmed control surfaces, which may be operated by dynamic pressure, electronics, etc., must also be calibrated. During the flight test program, it is frequently necessary to rig a control, such as the swashplate or tail rotor blade angle, to the allowable critical extreme of the tolerance band. For example, it would be necessary to rig the tail rotor to the minimum allowable blade angle if meeting the requirements of §27.143(c) would be in question. The same consideration must be given to all rotorcraft controls and movable aerodynamic surfaces where questionable compliance with the regulations may exist. If the rotor-induced vibration characteristics of the rotorcraft are significantly affected and require time-consuming rigging for such things as acceptable ride comfort, then the rotor(s) should be rigged to the allowable extreme tolerance limits to determine compliance, for example, with § 27.251.

(4) During the FAA/AUTHORITY flight test program, the crew should be especially alert for conditions requiring great attentiveness, high skill levels, or exceptional strength. If any of these features appear marginal, it is advisable to obtain another pilot's opinion and to carefully document the results of these evaluations. Section 27.141(b) provides the regulatory basis for these strength and skill requirements. The general requirements for a smooth transition capability between appropriate flight conditions are also included in § 27.141(b). These requirements must also be met during appropriate engine failure conditions for each category of rotorcraft.

(5) For night or IFR approval, § 27.141(c) contains the general regulatory reference, which requires additional characteristics for night and IFR flight. The appropriate flight test procedures are included in other portions of this advisory circular.

b. Procedures. none

AC 27.143. § 27.143 (Amendment 27-21) CONTROLLABILITY AND MANEUVERABILITY.

a. Explanation.

(1) This regulation contains the basic controllability requirements for normal category rotorcraft. It also specifies a minimum maneuvering capability for required conditions of flight. The general requirements for control and for maneuverability are summarized in § 27.143(a) which is largely self-explanatory. The hover condition is not specifically addressed in § 27.143(a)(2) so that the general requirement may remain applicable to all rotorcraft types, including those without hover capability. For rotorcraft, the hover condition clearly applies under "any maneuver appropriate to the type."

(2) Paragraphs (b) through (e), § 27.143, include more specific flight conditions and highlight the typical areas of concern during a flight test program.

(i) Section 27.143(b) specifies flight at V_{NE} with critical weight, center of gravity (CG), rotor RPM, and power. Adequate cyclic authority must remain at V_{NE} for nosedown pitching of the rotorcraft and for adequate roll control. Nosedown pitching capability is needed for control of gust response and to allow necessary flight path changes in a nosedown direction. Roll control is needed for gust response and for normal maneuvering of the aircraft. In the past, 10 percent control travel margin has been applied as an appropriate minimum control standard. The required amount of control power, however, has very little to do with any fixed percentage of remaining control travel. There are foreseeable designs for which 5 percent remaining is adequate and others for which 20 percent may not be enough. The key is, can the remaining longitudinal control travel at V_{NE} generate a clearly positive nosedown pitching moment, and will the remaining lateral travel allow at least 30° banked turns at reasonable roll rates? Moderate lateral control reversals should be included in this evaluation and since available roll control can diminish with sideslip, reasonable out of trim conditions (directionally) should be investigated. This “control remaining” philosophy must also be applied for other flight conditions specified in this section.

(ii) Section 27.143(c) requires a minimum control capability for hover and takeoff in winds of 17 knots from any azimuth. Control capability in wind from zero to at least 17 knots must also be shown for any other appropriate maneuver near the ground such as rolling takeoffs for wheeled rotorcraft. These requirements must be met from standard sea level conditions to the maximum altitude capability of the rotorcraft or 7,000 feet, whichever is less. On rotorcraft incorporating a tail rotor, efficiency of the tail rotor decreases with altitude so that a given sideward flight condition requires more pedal deflection, a higher tail rotor blade angle, and more horsepower. Hence, directional capability in sideward flight (or at critical wind azimuth) is most critical during testing at a high altitude site.

(iii) Section 27.143(d) requires adequate controllability when an engine fails. This requirement specifies conditions under which engine failure testing must be conducted and includes minimum required delay times.

(A) For rotorcraft which meet the engine isolation requirements of transport Category A, demonstration of sudden complete single-engine failure is required at critical conditions throughout the flight envelope including hover, takeoff, climb at V_Y, and high speed flight up to V_{NE}. Entry conditions for the first engine failure are engine or transmission limiting maximum continuous power (or takeoff power where appropriate) including reasonable engine torque splits. For multiengine Category A installations (three or more engines) subsequent engine failures should be conducted utilizing the same criteria as that used for first-engine failure. The applicant may limit his flight envelope for subsequent failures. Initial or sequential engine failure tests are ordinarily much less severe than the “last” engine failure test required by § 27.75(b). The conditions for last-engine failure are maximum continuous power, or 30-minute power if that rating is approved, level flight, and sudden engine failure with the same pilot delay of 1 second or normal pilot reaction time, whichever is greater.

(B) For rotorcraft without transport Category A engine isolation, demonstration of sudden complete power failure is required at critical conditions throughout the flight envelope. This includes speeds from zero to V_{NE} (power-on) and conditions of hover, takeoff, and climb at V_Y. Maximum continuous power is specified prior to the failure for the cruise condition. Power levels appropriate to the maneuver should be used for other conditions. The corrective action time delay for the cruise failure should be 1-second or normal pilot reaction time (whichever is greater). Cyclic and directional control motions which are apart of the pilot task of flight path control are normally not subject to the 1-second restriction; however, the delay is always applied to the collective control for the cruise failure. If the aircraft flying qualities and cyclic trim configuration would encourage routine release of the cyclic control to complete other cockpit tasks during cruise flight, consideration should be given to also holding cyclic fixed for the 1-second delay. Although the same philosophy could be extended to the directional controls, the likelihood of the pilot having his feet away from the pedals is much lower, unless the aircraft has a heading hold feature. Rotor speed at execution of the cruise condition power failure should be the minimum power-on value. The term “cruise” also includes cruise climb and cruise descent conditions. Normal pilot reaction times are used elsewhere. Although this requirement specifies maximum continuous (MC) power, it does not limit engine failure testing to MC power. If a takeoff power rating is authorized for hover or takeoff, engine failure testing must also be accomplished for those conditions. Following power failure, rotor speed, flapping, and aircraft dynamic characteristics must stay within structurally approved limits.

(iv) Section 27.143(e) addresses the special case in which a V_{NE} (power-off)is established at an airspeed value less than V_{NE} (power-on). For this case, engine failure tests are still required at speeds up to and including V_{NE} (power-on), and the rotorcraft must be capable of being slowed to V_{NE} (power-off) in a controlled manner with normal pilot reactions and skill. There is, however, no controllability requirement for stabilized power-off flight at speeds above 1.1 V_{NE} (power-off) when V_{NE} (power-off) is established per § 27.1505(c).

(v) Application of the controllability requirement for pitch, roll, and yaw at speeds of 1.1 V_{NE} (power-off) and below is similar to that described above for power-on testing at V_{NE}. Sufficient directional control must exist to allow straight flight in autorotation during all approved maneuvers including 30° banked turns up to V_{NE} (power-off) with some small additional allowance for gust control. Adequate controllability margins must exist in all axes throughout the approved autorotative flight envelope. Testing to V_{NE} at MC power per § 27.143(b), 1.1 V_{NE} at power for 0.9 V_H per § 27.175(b) or § 27.1505, and to 1.1 V_{NE} (power-off) in autorotation per § 27.143(e) should be sufficient to assure adequate control margin during a descent condition at high speed and low power. The high speed, power-on descent condition should be checked for adequate control margin as a “maneuver appropriate to the type.” There has been one instance where insufficient directional pedal was available to maintain a reasonable trimmed sideslip angle with low power at very high speeds, and a case where there was insufficient forward and lateral cyclic available to reach the power on V_{NE}. The insufficient directional pedal margin was due to the offset vertical stabilizers. The lack of cyclic stick margin was because the cyclic stick migrated to the right as power was reduced, and the control limits were circular. This provided less total available forward cyclic stick travel when the cyclic was moved right and forward about 45° from the center position. Each of the above rotorcraft was certificated with a rate of descent limitation to preclude operation in the control-limited area.

(vi) An evaluation of the emergency descent capability of the rotorcraft should be made, either analytically or through flight test. Areas of consideration are the rate of descent available, the maximum approved altitude, and the time before a catastrophic failure following the loss of transmission oil pressure or other similar failure. Each rotorcraft should have the capability to descend to sea level and land from the maximum certificated altitude within the time period established as safe following a critical failure. If the time period does not permit a sea level landing, the maximum height above the terrain must be specified in the limitation section of the Rotorcraft Flight Manual.

(3) The required controllability and maneuvering capabilities must also be considered following the failure of automatic equipment used in the control system (§ 27.672). Examples include stability augmentation systems (SAS), stability and control augmentation systems (SCAS), automatic flight control systems (AFCS), devices to provide or improve longitudinal static stability such as a pitch bias actuator (PBA), yaw dampers, and fly-by-wire elevator or stabilator surfaces. These systems all use actuators of some type, and are subject to actuator softover and hardover malfunctions. The flight control system should be evaluated to determine whether an actuator jammed in an extreme position would result in reduced control margins. Generally, if the flight control system stops are between the actuator and the cockpit control, the control margin will be affected. If the control stops are between the actuator and the rotor head, the control margins may not be affected, but the location of the cockpit control may be shifted. This could produce interference with other items in the cockpit. An example of this would be a lateral actuator jammed hardover causing a leftward shift in the cyclic stick position. Interference between the cyclic stick, the pilot's leg, and the collective pitch control could reduce the left lateral control available and reduce left sideward flight capability. In the case of fly-by-wire surfaces, both the high speed forward flight controllability and the rearward flight capabilities could be affected. Flight control systems that incorporate automatic devices should be thoroughly evaluated for critical areas. Every failure condition that is questionable should be flight tested with the appropriate actuator fixed in the critical failure position. These failures may require limitations of the flight envelope. Any procedure or limitation that must be observed to compensate for an actuator hardover and/or softover malfunction should be included in the Rotorcraft Flight Manual.

b. Procedures.

(1) Flight test instrumentation should include ambient parameters, all flight control positions, rotor RPM, main and tail rotor flapping (if appropriate), engine power instruments, and throttle position. Flight controls that are projected to be near their limits of authority should be rigged to the most adverse production tolerance. A very accurate weight and balance computation is needed along with a precise knowledge of the aircraft's weight/CG variation as fuel is burned.

(2) The critical condition for V_{NE} controllability testing is ordinarily aft CG, MC power, and minimum power-on rotor RPM, although power and RPM variations should be specifically evaluated to verify their effects. The turbine engine is sensitive to ambient temperatures which affect the engine's ability to produce rated maximum continuous torque. Flight tests conducted at ambient temperatures that cause the turbine temperature to limit maximum continuous power would not produce the same results obtained at the same density altitude at colder ambient temperatures where maximum continuous torque would be limiting. Forward CG should be spot checked for any "tuck under" tendency at high speed. The V_{NE} controllability test is normally

accomplished shortly after the 1.1 V_{NE} (or 1.1V_H) point obtained during stability tests required by § 27.175(b). Controllability must be satisfactory for both conditions. If V_{NE} varies with altitude or temperature, V_{NE} for existing ambient conditions is utilized for the test. Extremes of the altitude/temperature envelope should be analyzed and investigated by flight test.

(3) The critical condition for controllability testing in a hover is ordinarily forward CG at maximum weight with minimum power-on rotor RPM. For rearward flight testing of configurations where the forward CG limit varies with weight, low or high gross weight may be critical. Lateral CG limits should also be investigated. A calibrated pace vehicle is needed to assure stabilized flight conditions. Surface winds should be less than 3 knots throughout the test sequence. Testing can be done in higher stabilized wind conditions (gusting less than 3 knots); however, these conditions are very difficult to find and the method is very time consuming due to the necessity of waiting for stabilized winds. Testing in calm winds is preferred. Hover controllability testing should be accomplished with the lowest portion of the rotorcraft at the published hover height above ground level; however, the test altitude above the ground may be increased to provide reasonable ground clearance. Although the necessary yaw response will vary somewhat from model to model, sufficient control power should be available to permit a clearly recognizable yaw response after full directional control displacement when the rotorcraft is held in the most critical position relative to wind.

(4) Prior to engine failure testing, it is mandatory that the pilot be fully aware of his engine, drive system, and rotor limits. These limits were established during previous ground and flight tests and should be specified in the TIA. Particular attention should be given to minimum stabilized and minimum transient rotor RPM limits. These values must be included in the TIA and should be approached gradually with a build-up in time delay unless the company testing has completely validated all pertinent aspects of engine failure testing. On Category A installations, the maximum power output of each engine must be limited so that when an engine fails and the remaining engine(s) assume the additional load, the remaining engine(s) are not damaged by excessive power extraction and over-temping. This is needed for compliance with § 27.903(b). The propulsion engineer should have assured that this feature was properly addressed in the engine and drive system substantiation; however, it must be assumed that for some period of time the pilot may extract maximum available power from the remaining engine(s) when an engine fails during critical flight maneuvers. Substantiation of this feature should be accomplished primarily by engine and drive system ground tests.

(5) Longitudinal cyclic authority at V_{NE} with any power setting must permit suitable nosedown pitching of the rotorcraft. If the remaining control travel is considered marginal, tests should include applications up to full control deflection to assess the remaining authority. Some knowledge of the aircraft's response to turbulence is useful in assessing the remaining margin. As a minimum, the rotorcraft must have adequate margin available to overcome a moderate turbulent gust and must not have any divergent characteristic which requires full deflection of the primary recovery control to arrest aircraft motion. If other controls must be utilized to overcome

adverse aircraft motion, the results are unacceptable; e.g., if a pitch up tendency resulting from an actual or simulated moderate turbulent gust cannot be satisfactorily overcome by remaining forward cyclic, the use of throttle or collective controls to assist the recovery is not an acceptable procedure; however, the use of lateral cyclic to correct roll in conjunction with forward cyclic to correct pitchup is satisfactory. Obviously during the conduct of these tests, all available techniques should be utilized when the pilot finds himself "out of control." However, compliance with this section requires that recovery must be shown by use of only the primary control for each axis of aircraft motion.

(6) Cyclic control authority in autorotation must be sufficient to allow adequate flare capability and landing under the all-engine-inoperative requirements of § 27.75.

AC 27.151. § 27.151 (Amendment 27-21) FLIGHT CONTROLS.

a. Explanation. Excessive breakout or preload in the flight controls produces control system force discontinuities which result in increased workload and controllability problems for the pilot. Similarly, excessive freeplay results in lost motion which increases pilot workload and, in an extreme case, could lead to a hazardous pilot-induced oscillation. In some designs friction can provide a positive contribution to the function of the flight controls (e.g., masking aerodynamic feedback in reversible systems). At some point, friction will have a detrimental effect on the pilot's ability to properly control the machine. In the case of an irreversible design equipped with an artificial force feel system in pitch and roll, excessive friction can mask a shallow force gradient making positive stick centering and control force static stability difficult if not impossible to demonstrate. In such an instance, the initial choice of fixes might include implementation of a steeper force gradient or addition of a force preload. Unfortunately, these solutions often lead to the kinds of problems discussed earlier. Care must therefore be exercised during the initial design phase to ensure that the components and characteristics of the flight control system are well matched.

b. Procedures. Regardless of the flight control system sophistication, it is important that the test pilot understand the system configuration prior to flight evaluation. Appropriate mechanical characteristics should be documented. For VFR aircraft, the mechanical characteristics are typically assessed in flight on a qualitative basis. If a controllability or workload problem is identified, a more detailed investigation would be necessary. Since IFR certification rules include specific trim and force requirements, a more quantitative investigation of mechanical characteristics is normally conducted. The constantly varying feedback forces of reversible flight control systems generally make such designs unsuitable for IFR application. Irreversible system mechanical characteristics can often be partially documented on the ground with external hydraulic and electrical power supplies connected to the aircraft. Knowledge of the breakout, friction, and force gradient characteristics prior to flight can be useful to the pilot during flight evaluation of the system.

AC 27.161. <u>§ 27.161 (Amendment 27-21) TRIM CONTROL</u>.

a. <u>Explanation</u>.

(1) The pilot has many tasks to perform with each hand during sustained flight conditions. The trim requirement is intended to reduce the physical demands to maintain a given flight condition. It is not intended to require that control forces be reduced to zero by the trim control during dynamic maneuvers such as takeoff acceleration.

(2) A number of devices may be used to produce the necessary trim characteristics. One popular method of meeting this requirement is through the use of control balance springs in conjunction with a small amount of built-in control system friction. Other methods include use of friction, magnetic brakes, bungees, and irreversible mechanical schemes.

(3) This regulation is not intended to require zero friction or zero breakout force in the control system, nor is it intended to require automatic control recentering. The regulation, in fact, specifically prohibits excessive high friction or high breakout forces which would produce undesirable discontinuities in the primary control force gradient.

b. <u>Procedures</u>.

(1) If comprehensive company flight test data are available, compliance with this requirement can quickly be found by spot checking extreme center of gravity loadings. Trim tests can ordinarily be done during the course of other flight test activities. To conduct the test, briefly release the control at the required flight conditions and determine that the control does not move. The words "any appropriate speed" ordinarily include any speed from hover to V_H. If the control system trim device might be subject to temperature or humidity effects, these should be investigated at a minimum of two altitude extremes and during several test phases.

(2) If a pilot controllable variable friction device is incorporated, compliance with this requirement must be shown at the minimum adjustable value. The maximum value of adjustable friction should not completely lock the flight controls.

(3) Continued compliance with this requirement should be ensured through a production procedure. If minimum friction or centering springs are used, it is desirable for the manufacturer to include some adjustment capability for production differences. The explanation and procedures discussed here are applicable for VFR approval under § 27.161. For additional IFR trim requirements, refer to AC 27 Appendix B.

AC 27.171. <u>§ 27.171 STABILITY: GENERAL</u>.

a. <u>Explanation</u>. This section is intended to require a manageable pilot workload for the minimum crew under foreseeable operating conditions.

b. Procedures.

(1) Compliance with the requirements of this section can often be obtained for the VFR condition without any specific or designated flight testing. If the rotorcraft is marginal in regard to pilot strain and fatigue, the FAA/AUTHORITY pilot should be assured, through special tests if necessary, that the aircraft can be satisfactorily flown throughout the maximum endurance capabilities of the rotorcraft including night and turbulence conditions if those are critical. This test should be conducted with minimum required systems in the aircraft and with minimum flightcrew.

(2) Reasonable failure conditions which add to pilot workload, strain, and fatigue should be evaluated (electrical, hydraulic, and mechanical failures, etc.). The necessary times associated with flight with a failed system must be appropriate to the flight manual procedures for each failure. A failure condition requiring immediate landing would obviously require shorter evaluation time than a condition allowing continued flight to destination.

(3) IFR approvals necessitate a careful evaluation of paragraphs b (1) and (2) above. In IFR operations, weather conditions frequently necessitate continued flight to destination or diversion to alternate airports with critical failures. Immediate landing may not be feasible. The evaluating pilot must ensure pilot strain and fatigue are acceptable during typical flight profiles for each type of operation to be approved.

AC 27.173. § 27.173 (Amendment 27-21) STATIC LONGITUDINAL STABILITY.

a. Explanation.

(1) This rule contains control requirements for both stability and control. Paragraph (a) contains the basic control philosophy necessary for all civil aircraft. Forward motion of the cyclic control must produce increasing speeds, and aft motion must result in decreasing speeds. For rotorcraft, this is accomplished with throttle and collective held constant. Rotorcraft with either highly stable or highly unstable static longitudinal stability characteristics can typically comply with the basic requirement for control sense of motion. However, the intent and interpretation of this paragraph is to provide a stable stick position versus airspeed gradient. Therefore, a stabilized airspeed less than the trim speed requires a cyclic stick position aft of the trim stick position, and a stabilized airspeed greater than the trim speed requires a cyclic stick position forward of the trim speed stick position.

(2) The remainder of § 27.173, through reference to § 27.175, contains the basic control position requirements necessary to establish a minimum level of static longitudinal stability. Positive stability is found for conditions of climb, cruise, and autorotation in § 27.175 by requiring a stable stick position gradient through a specified speed range. A defined level of instability is permitted for the hovering condition.

b. Procedures.

(1) The control requirement of this section is so essential to basic flight mechanics that compliance may be found during conventional flight testing for compliance with other portions of the regulations. No special or designated testing should be required.

(2) The procedures necessary to ensure compliance with the stability requirements of this section are contained under § 27.175.

AC 27.175. § 27.175 (Amendment 27-21) DEMONSTRATION OF STATIC LONGITUDINAL STABILITY.

a. Explanation.

(1) This rule incorporates the specific flight requirements for demonstration of static longitudinal stability. Specific loadings, configurations, power levels, and speed ranges are stated for conditions of climb, cruise, autorotation, and hover.

(2) Some rotorcraft in forward flight experience significant changes in engine power with changes in airspeed even though collective and throttle controls are held fixed and altitude remains relatively constant. For these cases, the guidance in § 27.173 which states that throttle and collective pitch must be held constant is appropriate for administration of this rule, and the specified power in § 27.175(a), (b), and (c) should be considered as power established at initial trim conditions. This will result in slightly higher or lower torque readings at "off trim" conditions. Collective and throttle controls are held constant when obtaining data during climb, cruise, and autorotation tests.

(3) The effects of rotor RPM on autorotative static stability should be determined and positive stability demonstrated for the most critical RPM. Values for RPM can be expected to change as airspeed is varied from the "trimmed" condition. The manufacturer's recommended autorotation airspeed is ordinarily used for trim.

(4) Hovering is considered a flight maneuver for which the pilot repeatedly adjusts collective to maintain an approximately constant altitude above the ground. For hover stability tests, collective and throttle adjustments are made as necessary to maintain an approximately constant height above the ground. Also, a limited amount of negative longitudinal control travel is allowed with changes in speed.

b. Procedures.

(1) Instrumentation.

(i) Sensitive control position instrumentation is mandatory. Engine power parameters should be recorded at trim. For testing of minor modifications or when using a “before and after” method, a tape measure or a stick plotting board may be utilized. A stick plotting board consists of a level surface with a clean sheet of paper on it attached to the cockpit or seat structure. The installation must not interfere when the flight controls are fully displaced. A recording pencil is attached to the cyclic control by an offsetting arm in such a manner that it can be pushed down on the board to record relative cyclic position at key times during test maneuvers. The figure AC 27.175-1 plot is a typical presentation of longitudinal static stability.

(ii) Other necessary parameters include pressure altitude, ambient temperature, and indicated airspeed (pace vehicle or theodolite speed for hover tests). For hover tests, hover height (radar altitude if available) and surface winds should be documented. Two-way communication with a pace vehicle is highly desirable. Ground safety equipment is desirable.

(2) Ambient Conditions. Smooth air is necessary for stability testing. Allowable wind conditions for hover stability testing are the same as those for hover controllability tests. Extrapolation is covered in paragraph AC 27.45.

(3) Loading. Aft center of gravity (CG) is ordinarily critical for longitudinal stability testing, although high speed flight and hover should be checked at full forward CG and maximum weight. At aft CG, light or heavy weight conditions can be critical. The manufacturer’s flight data should be reviewed to determine critical loading conditions.

(4) Conducting the Test.

(i) The rotorcraft should be established in the desired configuration and flight condition (climb, cruise, autorotation) with the required power and rotor speed at the trim airspeed. The collective stick should be fixed in that position, usually by applying sufficient friction to ensure that it is not inadvertently moved. For autorotative tests, a rotor speed should be selected so that the variations in rotor speed as airspeed and altitude change do not exceed the allowable limits. This point is recorded as the trim point. Airspeed is then increased or decreased in about 10-knot increments, stabilizing on each speed and recording the data. At least two points on each side of the trim speed should be taken.

(ii) The cruise test should be accomplished by first determining V_H (level flight speed at maximum continuous power) at the test altitude. Then reduce power to establish a level flight trimmed condition at 0.9 V_H (or 0.9 V_{NE} if lower). This point is then recorded as the trim point. The collective pitch and throttle must remain fixed at the trim setting for the remainder of the test. The airspeed is then varied above and below the trim speed using the cyclic control to climb or dive slightly.

(iii) For climb and autorotation tests, conduct fixed collective tests through an altitude band (usually ±2,000 feet), first increasing airspeed as data points are collected, then decreasing speed through the same altitude band. It will probably not be possible to obtain the required data on one pass through the altitude band. If repeated passes are required, a trim point should be taken at the beginning of each pass unless very sensitive collective pitch position information is available in the cockpit. Generally, it will be possible to acquire all the high speed points on one pass and the low speed points on the second.

(iv) If extremely precise results are required, an alternate method of testing can be used to acquire the data at a constant altitude. For cruise, data can be obtained by alternating airspeeds above and below the trim speed to arrive in the vicinity of the test altitude as the point is recorded. This method results in very precise data because collective and throttle are not moved as airspeed is changed at a constant altitude. A typical sequence of speeds that could produce these results would be: 150 (V_H), 135 ($0.9V_H$) trim speed, 125, 145, 115, 155, 105, and 165.

(v) For rotorcraft with high rates of climb, a series of climbs, each at a different speed, may be required through a given altitude, utilizing sensitive instrumentation to ensure collective position is the same for each data point. In autorotation, a similar case arises and a series of descents, each at a different speed, may be required through a given altitude band, using sensitive instrumentation to ensure a repeatable collective position.

(vi) Hover tests should be conducted by maintaining an approximately constant altitude above the ground at the hover height established for performance purposes. The test altitude above the ground may be increased to provide reasonable ground clearance during rearward flight. Groundspeed is varied using a pace vehicle, theodolite, or other velocity measuring equipment. A pace vehicle is an aid in maintaining an accurate hover height. The pilot can accurately maintain height by controlling his sight picture of the pace vehicle (level with the roof, antenna, etc.). Hover stability tests are ordinarily conducted in conjunction with hover controllability tests because instrumentation and facilities are essentially the same.

(vii) Normally, climb, cruise, and autorotation tests should be conducted at low, medium, and high altitudes. See paragraph AC 27.45 for guidance on interpolation and extrapolation. High speed stability has been critical during cold weather testing. In two recent models, V_{NE} at cold temperatures has been limited by the stability requirements of § 27.175(b). Cold weather testing should be accomplished or a conservative approach for advancing blade tip Mach number should be used to limit cold weather V_{NE} to tip Mach number values demonstrated during warm weather testing.

(viii) Hover stability should be verified at low altitude and, if required, at high altitude. Refer to paragraph AC 27.45b(2) for guidance on expansion and extrapolation of altitude.

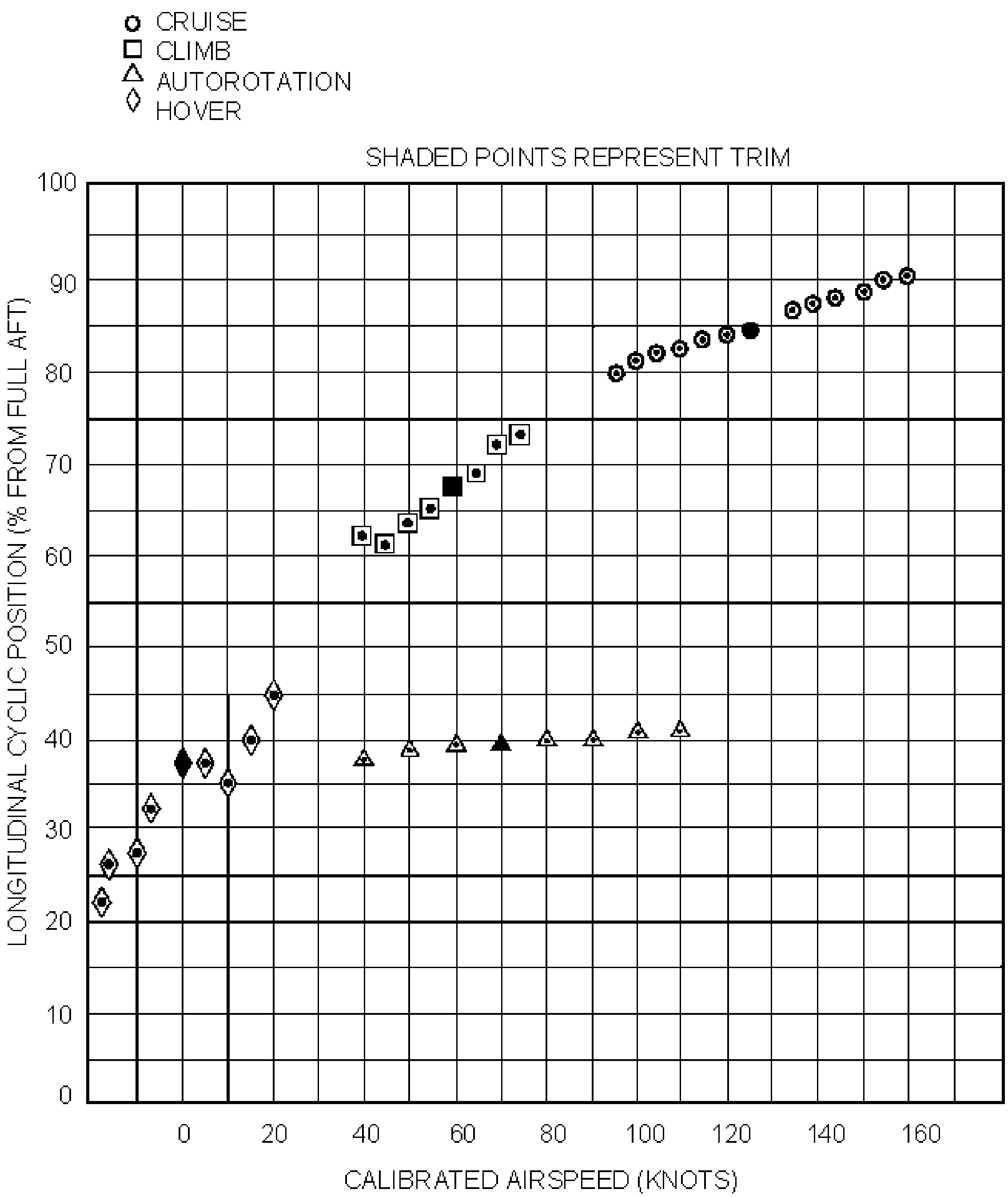


FIGURE AC 27.175-1 STATIC LONGITUDINAL STABILITY

AC 27.177. § 27.177 (Amendment 27-21) STATIC DIRECTIONAL STABILITY.

a. Explanation. This rule requires that positive static directional stability be demonstrated at the trim airspeeds defined in § 27.175. The trim speed for climb is V_Y and for cruise $0.9V_H$ or $0.9V_{NE}$ (whichever is less).

b. Procedures.

(1) Tests for static directional stability require instrumentation for pedal position and sideslip angle. To obtain accurate sideslip angle and airspeed information, a "yaw boom" is usually installed for the purpose of mounting a sideslip vane and swiveling airspeed pilot head outside the main rotor downwash region of influence. Special care should be taken to ensure that the yaw boom installation has been verified to be structurally adequate and free of dynamic instabilities for all combinations of airspeed and rotor speed likely to be experienced during the static directional evaluation. For some installations, the instrumentation yaw boom may influence the flying qualities of the rotorcraft itself. Thus, it is advisable to correlate yaw string displacement or slip indicator ball widths of skid with yaw boom sideslip angle, and then repeat a few critical points with the yaw boom removed.

(2) For some rotor system designs, the main and tail rotor flapping angle may be a critical instrumentation requirement for static directional testing. Both main and tail rotor flapping may increase dramatically at high airspeeds with increasing sideslip angle. Therefore, for rotor systems exhibiting this characteristic, flapping should be monitored carefully during the sideslip maneuver to avoid exceeding limitations. Static directional stability is normally defined in terms of pedal displacement required to maintain a steady heading sideslip. A single-rotor rotorcraft flying in coordinated flight will exhibit a small inherent sideslip due to tail rotor thrust and fuselage/main rotor sideforces. This condition is normally taken as trim with the inherent sideslip angle noted. Airspeeds should be the trim values described above. The procedures used to establish and maintain the steady heading sideslip can significantly influence test results. A generally accepted technique follows:

(i) Stabilize at the trim point, and note indicated airspeed.

(ii) Record trim conditions including inherent sideslip. Maintain fixed collective and throttle for the remainder of the maneuver.

(iii) Smoothly apply directional control and coordinate with lateral control to establish the desired sideslip angle. A steady track can best be ensured by maintaining a track over a straight landmark on the ground such as a section line or straight segment of powerline or highway.

(iv) Because drag increases with sideslip, the aircraft will decelerate. The trim airspeed should be maintained by entering a slight dive or decreased rate of climb. If a boom airspeed system is not used, a "standard" airspeed system will become excessively inaccurate at about 10° of sideslip. Rapidly yawing to the desired sideslip angle and noting and maintaining the new “standard” indicated airspeed may give adequate data. Control positions (directional as a minimum), sideslip angle, rotorspeed, airspeed, rate of descent, amount of ball deflection, and bank angle should be recorded. The pilot should note the physical sideforce feel experienced. The rule requires that sufficient cues accompany sideslip to alert the pilot when approaching sideslip limits. A minimum of two sideslip data points on each side of the trim point should be obtained to adequately define the slope of the pedal displacement versus sideslip angle relationship.

(v) Static directional stability plots can be expected to differ slightly on either side of the inherent sideslip angle. Positive static directional stability is indicated by increased left pedal displacement for a larger right sideslip and, conversely, increased right pedal for a larger left sideslip angle.

SUBPART B - FLIGHT

GROUND AND WATER HANDLING CHARACTERISTICS

AC 27.231. § 27.231 GROUND AND WATER HANDLING CHARACTERISTICS--GENERAL.

a. Explanation. The rule states: "The rotorcraft must have satisfactory ground and water handling characteristics, including freedom from uncontrollable tendencies in any condition expected in operation." In addition, §§ 27.235, 27.239, and 27.241 contain specific requirements concerning ground and water handling characteristic evaluations.

b. Procedures.

(1) During the flight test program and the F&R program (§ 21.35(b)(2)), the rotorcraft will be subjected to evaluations at various weight and CG conditions. Any uncontrollable tendencies found during these test programs must be corrected.

(2) Controllable or damped vibrations or oscillations on the ground or in the water are acceptable, provided the design limits of the rotorcraft are not exceeded.

(3) Any significant vibration or oscillation characteristics found during tests should be described in the test report, and the rotorcraft flight manual should contain appropriate descriptions and procedures to describe and either avoid or handle significant characteristics.

(4) For rotorcraft equipped with wheel gear, the evaluation should include takeoff, landing, and taxi at the maximum speed and at CG extremes. If a nose or tail wheel lock/swivel control is installed, each position should be evaluated for limiting takeoff, landing, and taxi speeds. Maximum substantiated speed values should be included in the RFM as limitations.

(5) For water operations, the wave height and frequency or "sea state" should be included as a limitation or, if no limit was reached during testing, the demonstrated values should be placed in the Performance Section of the RFM. Information or limits on the allowable "sea state" for rotor startup and shutdown should also be included.

AC 27.235. § 27.235 TAXIING CONDITION.

a. Explanation. The rotorcraft is designed for certain landing load factors (§§ 27.471 and 27.473). The rotorcraft must not attain a load factor in excess of the design load factor when taxied over the roughest ground that may reasonably be expected in normal operation at the expected taxi speeds. This rule applies to wheel landing gear equipped rotorcraft.

b. Procedures. The structural substantiation data contain the allowable design limits for the rotorcraft. A calibrated accelerometer or load factor “g” meter should be installed as near as practicable to the rotorcraft CG to record the maximum vertical load factor attained. Instrumentation of the landing gear and/or related structure may also be an acceptable means of showing compliance.

(1) Calibrated instrumentation should be installed to record the maximum loads or maximum vertical load factor attained during the taxi tests.

(2) The taxi surface should be evaluated for compliance with the rule. Corrugated surfaces as well as broken or uneven surfaces (in accordance with the rule) should be used.

(3) Representative typical taxi speeds, up to the maximum selected by the applicant, should be attained over the selected taxi surfaces.

(4) A light and heavy rotorcraft weight condition should be evaluated.

(5) Limitations appropriate for the rotorcraft design should be included in the flight manual. If these tests indicate that it is unlikely that limit load factors will be attained while taxiing, flight manual limitations may not be necessary.

(6) Pertinent taxi information obtained from these test conditions may be included in normal procedures of the flight manual.

AC 27.239. § 27.239 SPRAY CHARACTERISTICS.

a. Explanation. The intent of this requirement is to evaluate by demonstration that water spray does not obscure visibility (day or night) or damage the rotorcraft during normal waterborne operation (for those rotorcraft which have waterborne or amphibious capability).

b. Procedures.

(1) The following maneuvers should be evaluated in ambient conditions up to the proposed sea state or wave height for operation.

Con-fig.	Condition	Weight	CG	Rotor RPM	Alti-tude	Remarks
1	Taxi	Max	Optional	Max	SL	Speeds up to maximum proposed for water operation.
2	Hover	Max	Opt	Max	-	Determine critical hover height, if any.
3	Takeoff	Max	Opt	Max	SL	Unstick at maximum proposed water operation speed.
4	Land	Max	Opt	Max	SL	Touchdown at maximum proposed for water operation.
5	Shutdown	Opt	Opt	-	SL	Shut down the rotorcraft.
6	Start	Max	Opt	Max	SL	Start engines and release rotor brake.

(2) The maximum sea state or wave height evaluated under this rule should be stated and included in the limitations section of the flight manual.

(3) The effect of saltwater contamination and deterioration of turbine engines and other component parts of the rotorcraft should be considered in accordance with § 27.609 and paragraph AC 27.609. Information on saltwater effect and attendant corrective action should be provided in the flight manual, if appropriate, and in the maintenance manual.

AC 27.241. <u>§ 27.241 GROUND RESONANCE</u>.

a. <u>Explanation</u>.

(1) The rule states: “The rotorcraft may have no dangerous tendency to oscillate on the ground with the rotor turning.” This rule is a flight requirement that pertains to demonstrating freedom from dangerous oscillations on the ground. CAR Part 6, predecessor to FAR Part 27, originally contained a “strength requirement” under § 6.203 requiring ground vibration tests. These tests would identify critical vibration frequencies and modes of the rotorcraft. CAR Part 6, Amendment 6-4, effective October 1, 1959, removed this ground vibration requirement because the agency concluded that if any major component has a natural frequency which could be excited by some operating parameter, such a condition would be revealed in the course of other ground and flight tests. The FAA/AUTHORITY apparently was depending on demonstrations under § 6.131/§ 27.241 and the flight load survey data (§ 27.571) to satisfy the objective of the vibration test. However, Part 27, Amendment 27-2,

contained new § 27.663 adding reliability and damping action investigation requirements for ground resonance prevention means. A ground vibration survey was not reinstituted by the adoption of § 27.663. Compliance with § 27.663 does require investigation and substantiation as stated.

(2) "Ground resonance" is a mechanical instability of the aircraft while in contact with the ground, often when partially airborne. Stated another way "ground resonance" is a self-excited mechanical instability that involves coupling between the in-plane motion of the rotor blade and the motion of the rotorcraft as a whole on its landing gear (reference "Aerodynamic of the Helicopter," Gessow & Myers, page 308). It is caused by the motion of the blade in the plane of rotation (called in-plane vibration) coupled with a rocking or vertical motion of the aircraft as a whole. The tires, landing gear, and rotor pylon restraint structure act as a spring with a vibration frequency which coincides or couples with the natural in-plane frequency of the blade about a real or effective drag hinge in the plane of rotation. When the frequencies of the two motions (rotor and airframe) approach each other and couple, a violent shaking of the rotorcraft may occur which, if undamped, could result in the destruction of the rotorcraft.

(3) Ground resonance can occur due to flexibility in the rotor pylon restraint system as well as with landing gear flexibilities. This mode of vibration or resonance can happen in flight (called air resonance) as well as on the ground and should be addressed in the certification program. The evaluation should include variations in stiffness and damping that could occur in service to the rotor pylon restraints.

(4) Ground resonance may be prevented by placing the first order in-plane vibration frequency above the rotor turning speed.

(5) For such configurations which are not susceptible to ground resonance (first order in-plane frequency above rotor turning speed), a simple rotor RPM run-up and run-down with appropriate cyclic control displacement (i.e., excitation of any inherent vibrations) is adequate demonstration that a ground resonance condition does not exist. Unhinged "rigid" rotors, such as Bell Helicopter two-blade designs, are this type of rotor system.

(6) For configurations that are susceptible to ground resonance (i.e., first in-plane frequency is below the rotor turning speed), ground resonance is generally prevented by dampers on the blade acting in the plane of rotation, dampers on the landing gear (sometimes serving as oleo struts), or proper placement of the landing gear frequencies combined with rotor and/or landing gear dampers.

(7) Elastomeric components (in the rotor pylon support system, possibly in the landing gear, and possibly in the rotor head) are significantly affected by ambient temperature prior to warm-up. Their damping characteristics require thorough investigation for the range of rotorcraft operating environment as noted in § 27.663.

b. <u>Procedures</u>.

(1) Under all conditions, any oscillations which may be introduced should be damped. However, no instability should occur at any operating condition such as during RPM changes from minimum to maximum and idle to maximum. For rotorcraft with wheel gear, uneven taxi surfaces in conjunction with particular taxi speeds, may excite ground resonance and should be evaluated by taxiing on typical surfaces. This evaluation may be conducted in conjunction with the tests of § 27.235. In operation, the resonance characteristics should be checked during takeoff and landing at zero speed and during run-on landings using various power values.

(2) For those aircraft equipped with Stability Augmentation Systems (SAS), all ground resonance investigations should be conducted with SAS on and SAS off. This includes the hovering and running takeoffs and landings, taxi tests, and specific ground resonance tests noted herein. Consideration should be given to conducting tests in various SAS configurations such as roll channel on and pitch channel off, where such configurations are possible and authorized.

(3) For each rotorcraft configuration tested, the aircraft should be positioned on the ground in flat pitch with the rotor stabilized at the minimum practical rotational speed or optionally at a speed shown analytically to have significant margin from indicated resonant conditions. Control system inputs should be used to disturb the system for evaluation of subsequent damping.

(4) For each incremental increase in rotor speed and for each rotor speed setting at increments of collective pitch settings, cyclic and collective inputs should be investigated prior to proceeding to the next rotor speed setting. These inputs should cover the appropriate range and combinations of amplitude and frequency. The collective pitch setting increments should range from flat pitch to light on the landing gear prior to fully airborne, depending upon the test sequence for minimum risk.

(5) Cyclic pitch inputs should be made either by the pilot through the cyclic stick or through a signal-generating device working in conjunction with the cyclic controls. For each frequency of input, amplitude of the inputs should be increased incrementally and ultimately should be large enough to generate responses representative of normal ground and flight operation on the rotor and support system. The inputs should continue for a time sufficient to obtain representative responses, typically time sufficient to execute five complete circles of the cyclic stick (about neutral) at the selected frequency.

(6) The excitation frequency should be such as to excite the blade in-plane frequency. Rotor speed settings should be increased to 1.05 times the maximum power-on rotor speed. Collective pitch settings should be increased in increments of not more than 20 percent to maximum collective or alternately to the collective setting required to become partially airborne (when the cyclic is displaced as noted).

(7) Typically, articulated rotor aircraft have natural frequencies on the blade in lag of approximately 0.3 times the power-on main rotor RPM. Soft in-plane rotors have natural frequencies approximately 0.7 times the main rotor RPM. Therefore, for example, for a rotorcraft with an in-plane frequency of 0.3/rev, operating at 300 RPM, and with 6 inches of total lateral cyclic stick displacement, the stick should be rotated for 5 revolutions in a 0.6-inch-diameter circle at ((1-.03) x 300 RPM) or 3.5 cycles per second to attempt excitation of possible resonant frequencies. At the conclusion of the excitation, the cyclic stick should be returned to the neutral position while continuing the recording of data listed in paragraph b(13).

(8) The excitation process should include cyclic excitation inputs from the directional and longitudinal controls if critical for the type of rotorcraft being evaluated.

(9) If onset of ground resonance is encountered, one possible corrective action is to increase the collective pitch and rotor speed and become airborne. However, lowering the collective pitch and applying the rotor brake (if installed) or rolling off the throttles has been effective for some designs and is considered a satisfactory procedure if resonance can be consistently stopped.

(10) With the rotor speed stabilized, landing should be made at a touchdown speed which minimizes risk.

(11) Special Considerations.

(i) The influence of variables, including environmental effects, corresponding aircraft component characteristic changes, operational parameters, and surface conditions should be investigated over the ranges proposed for certification. Additionally, the potential of misservicing and possible failure modes should be evaluated. For ground resonance qualification, where practical, variations from the baseline test configuration may be accomplished by ground run (§ 27.663(b) requires investigation of probable ranges of damping), analyses, component tests, aircraft shake test, the specification of special operational procedures in the rotorcraft flight manual, or a combination thereof. Detailed and rational analyses showing acceptable correlation to the baseline tests, and for which the input parameters were verified by drawings, calculations, component static or dynamic tests, or by aircraft shake tests simulating the conditions/configurations in question, may be used to limit testing to only those variables and operational conditions showing marginal or unacceptable system damping. All operational limitations should be clearly stated in the rotorcraft flight manual. A report of the analytical results and/or test results should be submitted per § 27.663.

(ii) Potential instability while airborne, called “air resonance,” may occur due to the dynamic coupling of the rotor flexibility and the pylon restraint flexibility. The same considerations apply to air resonance as to ground resonance except that the pylon restraint variables replace the landing gear variables. Air resonance should be addressed in the certification program.

(iii) When operating on the ground, there may be a tendency for the aircraft to exhibit a “ground bounce.” For many configurations, this is a benign, although undesirable phenomenon which may be aggravated by pilot induced oscillations (PIO), particularly if there is little or no friction on the collective.

(12) Rotorcraft with fully articulated rotor heads and landing gear oleos in either skid or wheel configuration have tendencies for ground bounce to occur when light on the oleos, either just prior to takeoff, just after landing contact, or during a power assurance check. This bounce may induce ground resonance, particularly if the intensity of the bounce is aggravated by PIO. The corrective action is either to lift off to a hover or to positively lower the collective and remain on the ground..

(13) Instrumentation and Data Acquisition.

(i) Atmospheric Conditions (to be manually noted):

(A) Altitude.
(B) OAT.
(C) Wind velocity.

(ii) Aircraft Configuration (to be manually noted):

(A) Gross weight.
(B) C.G.
(C) Tire pressure.
(D) Landing gear oleo pressure.

(iii) Instrumentation (for recording during test).

(A) Main rotor RPM.
(B) Time history of cyclic control fore-and-aft and lateral stick position.
(C) Time history of collective control stick position.
(D) Time history of rotor damper motion.*
(E) Time history of pylon component motion.*
(F) Time history of landing gear (oleo) motion.*
(G) Time history of aircraft motions.*

*As required to obtain modal damping

SUBPART B - FLIGHT

MISCELLANEOUS FLIGHT REQUIREMENTS

AC 27.251. § 27.251 VIBRATION.

a. Explanation.

(1) Each part of the rotorcraft must be free from excessive vibration under each appropriate speed and power condition (rule statement).

(2) This flight requirement may be both a qualitative and quantitative flight evaluation. Section 27.571(a) contains the flight load survey requirement that results in accumulation of vibration quantitative data. Section 27.629 generally requires quantitative data to show freedom from flutter for each part of the rotorcraft including control or stabilizing surfaces and rotors.

(3) Review Case No. 70 (reference FAA Order 8110.6) contains a policy statement concerning compliance with this rule. This policy statement is condensed here for convenience:

"The rotorcraft must be capable of attaining a 30° bank angle (turn), at V_{NE}, with maximum continuous power (maximum continuous torque) without encountering excessive roughness/vibration. The FAA/AUTHORITY requires the maneuver demonstration to provide the pilot with some maneuver capability at V_{NE} and further to provide the pilot some margin away from roughness when operating in turbulence." (This maneuver may result in a descent or a climb.)

(4) Section 27.1505 pertains to V_{NE} determination. Section 27.1509 pertains to rotor speed limits determination.

b. Procedures.

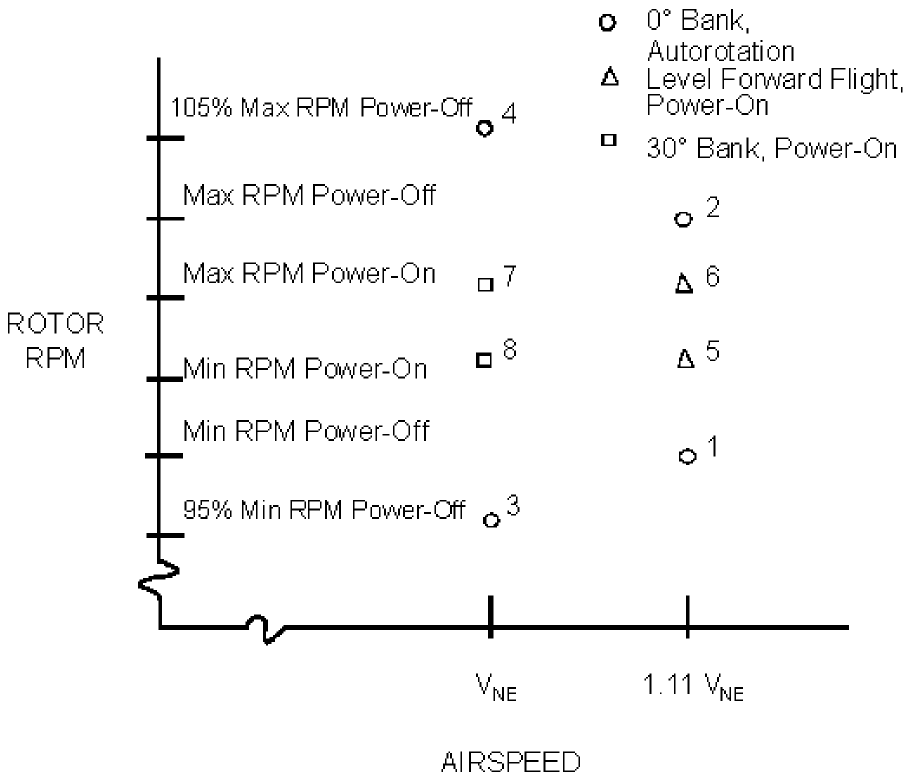
(1) During the company flight test program, the rotorcraft is flown to the appropriate rotor and airspeed limits at several weights to prove that the rotorcraft is free from excessive vibration under appropriate speed, power, and weight conditions. The flight loads survey quantitative data (reference § 27.571) and the applicant's qualitative and quantitative flight test data must also prove compliance with the requirement prior to issuing an authorization for official FAA/AUTHORITY flight tests.

(2) The flight load survey data obtained under § 27.571(a) will contain measured data concerning proof of freedom from flutter and excessive vibration. Pertinent critical flight conditions will be reinvestigated during FAA/AUTHORITY flight tests. The specific condition or conditions necessary to demonstrate compliance with § 27.251 vary with the rotorcraft design and with the minimum and maximum rotor

speeds, V_{NE} and V_D speeds, and weight and CG position. An illustration of the speed and RPM demonstration is shown in figure AC 27.251-1. (Also see paragraph AC 27.251b(4).)

(3) The airspeed and rotor speed limits investigated and established under §§ 27.33, 27.1503, 27.1505, and 27.1509 are also investigated and made a matter of record in the flight loads survey data. During the official FAA/AUTHORITY/TIA flight tests, critical parts of the rotorcraft may have limited instrumentation to reinvestigate and confirm that the critical conditions investigated during the flight load survey are satisfactory and do not result in excessive vibration. Use of instrumentation is optional if the flight loads data are conclusive.

(4) FAA/AUTHORITY policy for certification (Review Case No. 70) requires a “rotor roughness” flight demonstration of a 30° bank angle left and right at maximum continuous power (MCP) (maximum continuous torque which may be in excess of the maximum continuous temperature limit) at V_{NE}. To provide the pilot with some margin from roughness, the FAA/AUTHORITY requires maneuver demonstrations of 30° banked turns at V_{NE} without encountering excessive roughness. The maneuver should be conducted with the rotor speed at the minimum RPM and maximum RPM limits. During the flight load survey, this condition should be investigated and data recorded to ensure hazardous loads are not encountered for this “unusual” condition. As indicated, the flight condition will be reinvestigated during the FAA/AUTHORITY flight tests. See paragraph AC 27.251b(2) for illustration of this speed and RPM demonstration.



1. Autorotation at 1.11 $V_{NE\ (AR)}$ at minimum placard rotor speed.
2. Autorotation at 1.11 $N_{NE\ (AR)}$ at maximum placard rotor speed.
3. Autorotation at $N_{NE\ (AR)}$ at power-off minimum design limit rotor speed.
4. Autorotation at $N_{NE\ (AR)}$ at power-off maximum design limit rotor speed.
5. Forward flight 1.11 V_{NE} at minimum power-on rotor speed.
6. Forward flight 1.11 V_{NE} at maximum power-on rotor speed.
7. Right and left turn at V_{NE} at maximum power-on rotor speed with 30° bank angle.
8. Right and left turn at V_{NE} at minimum power-on rotor speed with 30° bank angle.
   Note: $V_{NE\ (AR)}$ may be less than V_{NE} .

FIGURE 27.251-1 DEMONSTRATION POINTS